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NASA Contract NAS 5-302

PROJECT APOLLO

A Feasibility Study of an Advanced Manned Spacecraft and System

FINAL REPORT

VOLUME V. HUMAN FACTORS
Book 1

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GENERAL ELECTRIC

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CHAPTER I

LIFE SUPPORT CONSIDERATIONS

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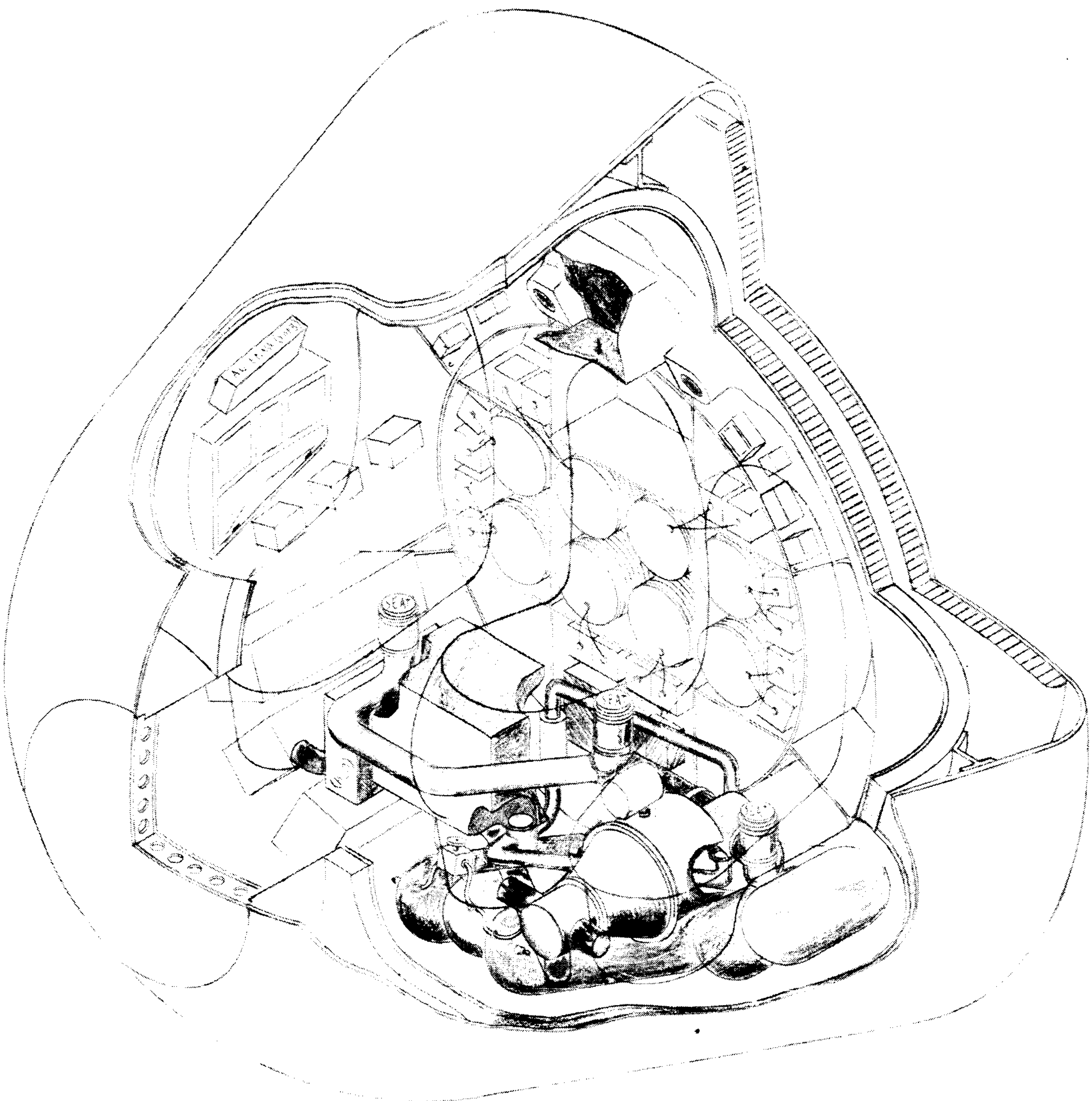

Cutaway view of re-entry vehicle showing equipment packaging

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I. LIFE SUPPORT CONSIDERATIONS

1.0 Introduction

The scientific potential of Project APOLLO — even the assurance of mission completion — is highly dependent upon human capabilities. Man's capacities allow him to act as a sensor, a variable-programmed general purpose computer, a data-reducer, a memory bank and scores of other such devices. These capacities will permit the flexible efficient performance of the scientific experiments and reconnaissance tasks which are the essence of the APOLLO mission and will permit the diagnosis, maintenance and modification of the on-board equipment necessary for survival and experimentation as well. At the same time, the presence of man penalizes a space vehicle system through constraints and weights imposed by the requirements for life support and for protection from extreme environmental stress. The Life Support and Human Factors study for Project APOLLO has been directed, therefore, at the determination of the boundary conditions for manning the APOLLO vehicle system and the demonstration of feasible hardware solutions.

In order for man's capabilities to be used to the fullest an appropriate environment must be provided during all phases of the mission. The broad range of requirements is met in terms of an unencumbered crew (the "shirt sleeve" environment) so that maximum human performance efficiency can be achieved when it is most critically needed. The study has resulted in the selection of a 7.0 psia gaseous atmosphere containing nitrogen as a diluent. Feasible methods for its maintenance and control are described as well as engineering solutions for the control of the environmental temperature, humidity and atmospheric contaminants. The physical stresses on the crew associated with launch, re-entry and abort are analyzed and methods of crew seating and restraint during periods of acceleration, vibration and noise are described, as well as means of protection in the event of decompression. Solutions to the problems of food, water, waste and personal hygiene and equipment are presented.

The significant problems of prolonged weightlessness and radiation have received special emphasis. Analysis has indicated that the physiological effects of prolonged weightlessness can be successfully handled by physiological methods without recourse to artificial gravity produced by rotation. Orbital experimental designs and bioinstrumentation emphasize the study of this problem. The radiation threat has been analyzed, and a feasible design approach described which includes APOLLO trajectory analysis for calculation of integrated radiation dose and for the evaluation of shielding requirements and secondary radiation within the vehicle during solar flares.

An intensive study has been carried out in the area of man-machine integration for the APOLLO mission and vehicle. The mission was broken down into a series of required control events and environmental characteristics and life support provisions were analyzed with respect to operator performance and tolerance. Tasks were allocated to man or machine based on criteria of optimizing system performance by the judicious use of man as a system component contributing to hardware simplification and reliability. Tasks were assigned to specific operators on the basis of work load and efficient overall time schedule, and crew size was verified. From such procedures the required control-display system and information requirements were developed and selection and training procedures and equipment were derived. Detailed and comprehensive considerations of the basic factors affecting sustained performance in the APOLLO system are given in the report and certain elements of the display and control system were executed in mockup form.

2.0 Atmosphere Selection and Maintenance

2.1 PHYSIOLOGICAL REQUIREMENTS

Because man normally spends his lifetime in a pressure environment of 1 atmosphere containing approximately 21 percent oxygen and 79 percent nitrogen, it may logically be assumed that he should be provided with the same ambient conditions when placed in a sealed space cabin such as APOLLO. From an engineering standpoint, however, consideration should be given to compromising man's natural environment without having to compromise his well-being. In order to resolve this bioengineering problem, consideration must be given to various pressures and gaseous compositions and their biological effects.

Oxygen

In order to establish a minimum value for the partial pressure of oxygen (pO_2), it is necessary to consider two cases: 100 percent oxygen and oxygen plus an inert gas.

2.1.1.1 100 PERCENT OXYGEN

A pCO_2 of 0-8 mm Hg in ambient air is low enough so that it can be neglected and thus, the alveolar O_2 partial pressure ($P_{A_{O_2}}$) may be calculated as follows:

$$P_{A_{O_2}} = P_B - P_{A_{CO_2}} - P_{A_{H_2O}} \qquad (1)$$

where P_A refers to alveolar pressure in mm Hg and P_B = barometric pressure.

Assuming no physiological adjustment to compensate for hypoxia ($P_{A_{O_2}} \cong 100$ mm Hg) and considering normal values for $P_{A_{CO_2}}$ and $P_{A_{H_2O}}$, then $P_B = 100 + 40 + 47 = 187$ mm Hg. This value may be taken as minimal for design purposes. Since a degree of hypoxia equivalent to breathing air at 10,000 feet ($P_{A_{O_2}} = 60$ mm Hg) may be tolerated for indefinite periods depending on the adaption capacity of an individual, the design minimum total pressure can be reduced to 147 mm Hg. Any decrease in this value will result in a proportionate decrease in the time of useful consciousness

(TUC) approaching the lung-heart-brain circulation time of about 8 to 12 seconds (P_B = 87 mm Hg).

2.1.1.2 OXYGEN PLUS AN INERT GAS

When some inert gas such as nitrogen, argon, helium, etc., is mixed with oxygen, the situation becomes somewhat more complicated than for a 100 percent-oxygen system. A minimum pO_2 for all total pressures, cannot be specified. However, for a series of ambient pressures ranging from one to 1/5 atmosphere as shown in Figure I-2-1, the minimum $P_{A_{O_2}}$ may be obtained for any given total pressure. To obtain these values, it has been assumed that the alveolar pCO_2 ($P_{A_{CO_2}}$) and alveolar respiratory exchange (R_A) remain constant at 40 mm Hg and 0.82, respectively. Thus, the alveolar pO_2 ($P_{A_{O_2}}$) may be calculated from the equation of Fenn and co-workers:

$$P_{A_{O_2}} = F_{I_{O_2}} (P_B - P_{A_{H_2O}}) - P_{A_{CO_2}} \left[F_{I_{O_2}} + \frac{(1-F_{I_{O_2}})}{R_A} \right] \qquad (2)$$

where $F_{I_{O_2}}$ = fractional O_2 concentration in dry inspired gas.

Assuming that the lowest acceptable $P_{A_{O_2}}$ is 100 mm Hg, then the lowest $P_{I_{O_2}}$ (inspired pO_2) ranges from 186 mm Hg for a 1/4 atmosphere system to 157 mm Hg for a 1 atmosphere system. These values are proportionately reduced if a $P_{A_{O_2}}$ of 60 mm Hg is taken for emergencies (see Figures I-2-1 and I-2-2).

It is of interest to note that the addition of CO_2 or H_2O in the inspired air moves the curves of Figure I-2-2 to the left. Figures I-2-1 or I-2-2 indicate that a slight safety factor remains in that some CO_2 (0–8 mm Hg) and H_2O (10 ± 5 mm Hg) will be present in the gaseous environment.

The maximum $P_{I_{O_2}}$ for any system for an indefinite period has been calculated by Mullinax and Beischer to be 425 mm Hg, whereas 760 mm Hg may be taken as a 12-hour maximum.

A problem that must be eventually solved is that of atelectasis produced by breathing atmospheres devoid of or having greatly reduced concentrations of an inert gas. During the past 20 years, much speculation and scattered pieces of evidence concerning

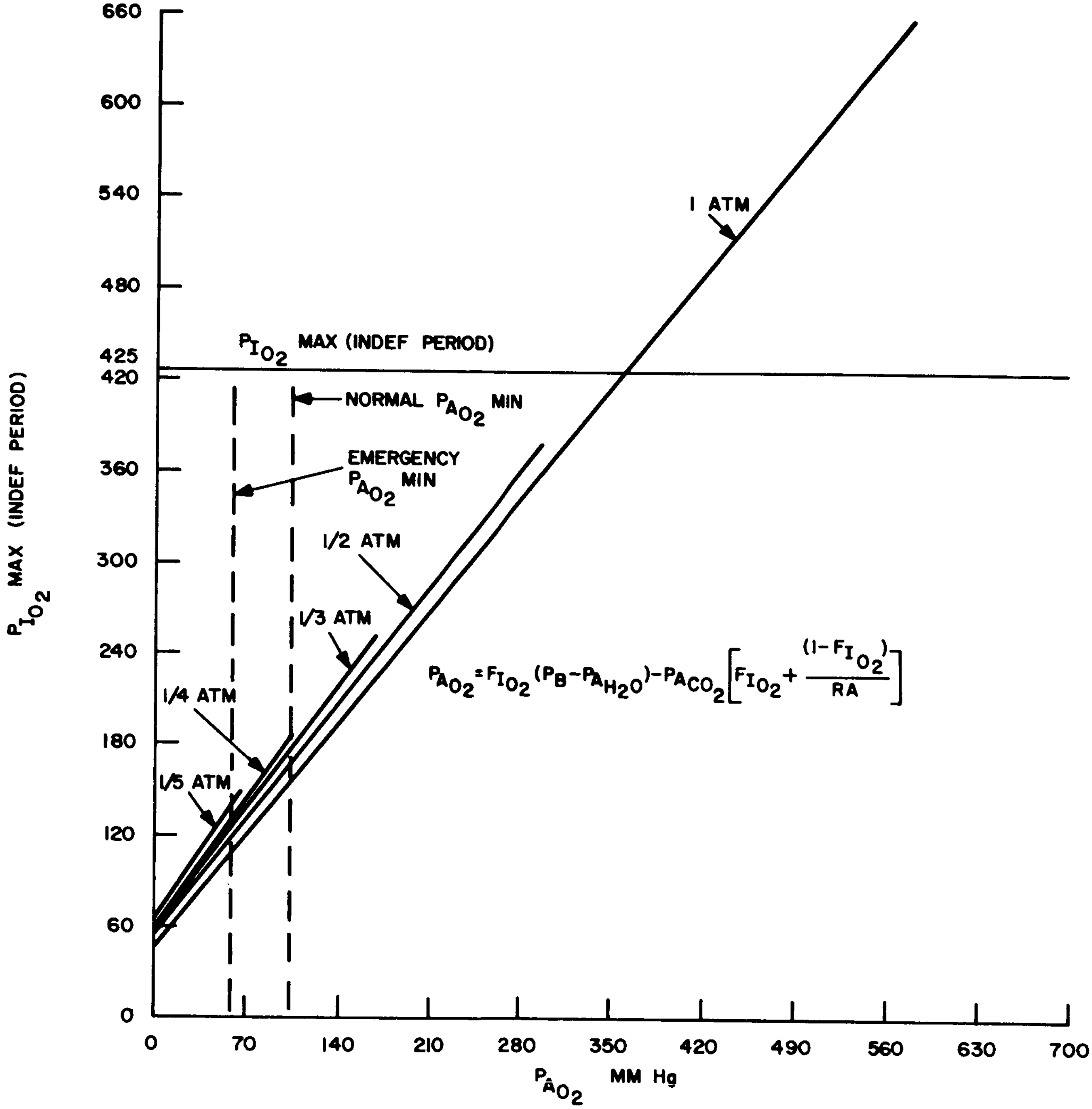

Figure I-2-1. Minimum $P_{A_{O_2}}$ for given total pressure

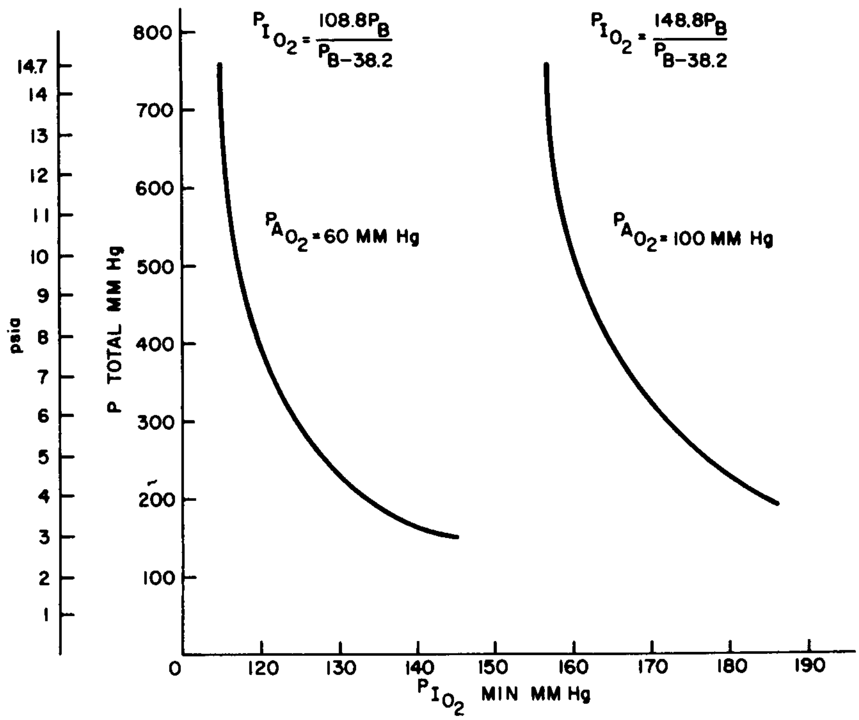

Figure I-2-2. Total pressure versus minimum partial pressure of oxygen plus inert gas

atelectasis in subjects breathing 100 percent oxygen at various pressures has appeared in the literature. Lambertsen has summarized some of this literature and given a theoretical basis for atelectasis. In a person breathing 100 percent oxygen, there will be a pressure differential between the alveolar gas and the arterial blood due to the normal "shunting" of mixed venous blood across the lungs without exposure to alveolar gas and to the mixing of the shunted venous blood with blood from the pulmonary capillaries. This pressure differential amounts to approximately 30 mm Hg while breathing oxygen at 760 mm Hg and about 5 mm Hg while breathing oxygen at 187 mm Hg. Furthermore, there is a Δ P of greater than 600 mm Hg for 760 mm Hg P_{IO_2} and about 50 mm Hg for 187 mm Hg P_{IO_2} between the aveoli and venous (internal juglar) blood. Thus, if a bronchiole becomes obstructed, all of the alveolar spaces distal to the obstruction would transfer their gases (O_2, CO_2, H_2O) to adjacent tissues and eventually collapse. If such an irreversible diffuse atelectasis occurs, the end result would be arterial anoxemia due to excessive shunting of blood across the lungs via those pulmonary capillaries supplying collapsed alveoli. It should be pointed out that the rate of alveoli collapse is not only a function of the oxygen fraction in the alveolar gas but also of the capillary blood flow. Dale and Rahn have demonstrated that the time to collapse for an obstructed lung of a dog breathing air at 1 atm. is 370 minutes. While breathing oxygen at 1 atm. the time to collapse was 6 minutes and only 1 minute while breathing oxygen at 200 mm Hg. It appears that nitrogen has an important function in "braking" lung collapse. Rahn has also reported that mice kept at 200 mm Hg. are able to survive indefinitely if they can make it past the first 24 hours. Here is a less drastic example of atelectasis than complete obstruction of one lung. Removal of the nitrogen "brake" apparently constitutes a stressing situation to which adaptation must occur. Atelectasis probably becomes more of a problem during sleep when the lungs are less active.

2.1.1.3 CARBON DIOXIDE AND WATER VAPOR

There is no evidence to indicate a permissible lower level of carbon dioxide (PI_{CO_2}). The anesthetic level of CO_2 is about 75 mm Hg. Thus, an upper limit between 0 and 75 mm Hg must be chosen. An early submarine level of 3 percent at 1 atmosphere was used for a number of years. However, Schaefer has demonstrated that 3 percent of CO_2 produces a biphasic excitation — depression reaction in humans. He suggested that a lower limit of 0.5 to 1 percent be used for CO_2. It would, therefore, appear safe to consider a CO_2 range of 0 to 8 mm Hg.

Water vapor limits will be discussed in detail in Appendix D. The suggested range is 10 ± 5 mm Hg.

2.1.1 Inert Gases

A summary of the use of inert gases in sealed cabins has been made by Roth. Six physiologically inert gases may be considered. These six gases are helium, neon, nitrogen, argon, krypton, and xenon. The narcotic effects of helium, neon, argon and nitrogen occur only at pressures considerably greater than one atmosphere. Since space cabins will probably not have total pressure greater than 1 atmosphere, it appears that no upper partial pressure limit need be assigned these gases. Krypton and xenon, however, owing to their high fat solubility, may be expected to show some narcotic action at pressures of 1/2 to 1 atmosphere. Cullen and Gross have compared xenon to ethylene in anesthetic capacity and noted the similarity in solubility coefficient (about 1.5) in oil. Since fat solubility and the molecular weight of the six aforementioned gases are directly related (see Table I-2-1), they may be ranked in order of decreasing anesthetic action:

$$Xe > Kr > A > N_2 > Ne > He.$$

As mentioned in the previous section on "Oxygen," the determination of a minimum concentration of an inert gas awaits further experimentation.

TABLE I-2-I. INERT GASES AND PROPERTIES [(a)]

Property	He	GAS				
		Ne	A	Kr	Xe	N_2
Atomic number	2	10	18	36	54	7
Molecular weight	4.00	20.18	39.93	82.92	130.2	28.02
Density (gm/liter)	0.178	0.889	1.78	3.71	5.585	1.25
Boiling point (deg C.)	-268.8	-245.9	-185.8	-151.8	-106.9	-195.8
Solubility coefficient in water, 38 C.	0.0086	0.0097	0.026	0.045	0.085	0.013
Solubility coefficient in oil, 38 C.	0.018	(0.046)*	0.14	0.43	1.7	0.067
Oil-water solubility ratio	1.7	(4.7)**	5.3	9.6	20.0	5.2
Relative diffusion through gelatin, 23C.	1.0	(0.45)***	0.30	0.21	0.13	0.36

* - Interpolated from Molecular Weight Factors.

** - Calculated from this table.

*** - Calculated from factors of Grahams' law.

(a) - After Roth (1959)

2.1.2.1 DECOMPRESSION SICKNESS

As the result of an emergency (or programmed) decrease in cabin atmospheric pressure, gases contained in solution within the body will rapidly or slowly leave the body via the lungs depending on the particular gas and its accessibility to the exterior. The gases within the body that cannot rapidly equilibrate with the cabin pressure will form bubbles within various tissues. If bubbles of sufficient number or size are formed, then pain, neurological effects, and possibly death may result. An analysis of factors controlling the rate of growth and decay of tissue bubbles during decompression in an idealized model has been made by Roth. He reported that the gas factor (solubility in oil^2 x diffusion coefficient in oil/solubility in H_2O) determined the maximum size of bubble and symptoms. (See Table I-2-II.)

TABLE I-2-II. DECOMPRESSION BUBBLE FACTORS[a]

Factor	He	Ne	A	Kr	Xe	N_2
Solubility in H_2O, 38 C. X relative diffusion in gelatin, 23 C.	0.0080	0.0044	0.0078	0.0095	0.011	0.0047
Solubility in oil, 38° C. X relative diffusion in gelatin, 23 C.	0.0180	0.0210	0.0420	0.0900	0.220	0.0240
Bubble factor	0.0310	0.0990	0.2200	0.8600	4.400	0.1300
Relative bubble factor nitrogen = 1	0.2400	0.7600	1.7000	6.6000	34.000	1.0000

(a) After Roth (1959)

These problems may be considered significant if the cabin pressure becomes less than about 5.0 psia as in the case of an atmosphere containing N_2 as an inert gas during a decompression from approximately sea level pressure. However, if a decompression protection device is provided capable of maintaining a pressure greater than 5.0 psia, decompression sickness should not prove to be a problem. If, on the other hand, the decompression protection device is capable of only maintaining a total pressure less than 5.0 psia, bends may occur if decompression is from sea level pressure. The bends problem in devices of less than 5.0 psia may be handled in two ways:

2.1.2.1.1 Maintenance of the Normal Cabin Pressure at Levels Less Than 1 Atmosphere

Marbarger and co-workers have shown that the incidence of bends at 2.99 psia (38,000 feet) for 2 hours is 6 percent if the subjects breathed 100 percent oxygen at sea level for 2 hours prior to decompression. There were no symptoms during the first hour. The amount of nitrogen eliminated in 2 hours was estimated by Marbarger to be 1051 cc (STP). If we assume the total nitrogen to be 1500 cc, then the bends which developed must have been due to the bubbles formed by the remaining 449 cc. Had the denitrogenation time been longer, it is probable that the incidence of bends would have been less. Bateman reported that a group of seven subjects who pre-breathed oxygen for 4 hours remained at 38,000 feet for 12 hours. A 4-hour denitrogenation should leave approximately 150 cc of nitrogen available for bubble formation.

If we assume a normal cabin pressure of 1/2 atmosphere with a 50-50 nitrogen-oxygen mixture and allow sufficient time for body equilibration at this pressure, 442 cc (STP) of nitrogen would remain in the tissues. This situation is comparable to breathing 100 percent O_2 at sea-level pressure for 2 hours. Thus, rapid decompression to 3 psia under these conditions may produce bends. For these circumstances, a normal cabin atmosphere of lower total pressure, decreased nitrogen content or decompression protection at greater than 3 psia must be provided. A 3.6-psia decompression protection pressure appears sufficient and would provide the occupant with a $P_{A_{O_2}}$ of 100 mm Hg.

The "hitch" that occurs in this method of bends protection is the possibility of decompression before equilibration at cabin pressure has occurred in the early phases of the flight. Denitrogenation prior to launch thus becomes mandatory for bends protection unless a decompression protection pressure of about 5 psia is used.

2.1.2.1.2 Use of An Inert Gas Other Than Nitrogen

Available evidence concerning other gases is sufficient to make an estimate of their usefulness in bends prevention. It would appear (Table I-2-II) that xenon, krypton, and argon are worse than nitrogen, neon is slightly better than nitrogen and helium is about four times better than nitrogen. Helium is obviously the inert gas of choice for the avoidance of decompression symptoms.

Another possible reason for the use of helium may be the prevention of atelectasis during acceleration. Due to its low density it will pass into alveoli through constricted bronchioles and perhaps prevent collapse. This condition may be likened to the alleviation of labored breathing in an asthmatic patient by the use of a helium-oxygen mixture. In view of the lack of data concerning helium, however, it would appear wise at the present to consider nitrogen as the inert gas of choice.

2.2 SELECTION OF THE ATMOSPHERE

2.2.1 Ground Rules for Atmospheric Selection

The "ground rules" used in selecting the internal atmosphere of the APOLLO vehicle are:

(a) The atmosphere must, in accordance with the "shirt-sleeve environment" philosophy, produce minimum physiological stress to the crew.

(b) In the event the vehicle is punctured, it should provide a viable atmosphere for as long as required for corrective action and, in the event the pressure vessel integrity is lost and it becomes necessary to maintain the crew at a reduced pressure by means of a secondary pressurization system, then the total pressure change should not cause aeroembolism.

(c) The atmosphere should not, insofar as possible, increase fire hazard by increasing the rate of combustion.

(d) The atmosphere selected should be one which is capable of being maintained with a minimum weight and volume system, requiring minimum electrical power.

Unfortunately, some of these objectives tend to place contradictory requirements upon various parameters and, therefore, it becomes necessary to trade off one objective against another.

2.2.2 Some Considerations of Oxygen Partial Pressure

The most essential ingredient in the atmosphere is oxygen. Gross permissible limits have been placed upon the oxygen partial pressure, the lower limit being the one which just maintains the alveolar pO_2 above the hypoxic level and the upper limit being the one which, if exceeded, will result in oxygen toxicity. Within this range, various factors must be traded off. The possibility of atelectasis, especially during acceleration, the fact that the oxygen leakage weight is a linear function of its partial pressure, and the sharp increase in combustion rate with oxygen partial pressure, all suggest that the pO_2 be kept close to the lower permissible level. Contradicting this is the desire to maintain as high a pO_2 as possible in order to provide the maximum useful time to the astronauts to take whatever action is indicated in the event of rapid leakage due to meteorite puncture or some component or structural failure.

The low probability of pressure vessel structural failure, coupled with the encouraging results of the meteoric puncture probability analysis and the readily accessible cocoon-type secondary pressurization system, indicate that the "time available in the event of rapid leakage" consideration does not outweigh the other factors which make it desirable to select a pO_2 close to the lower permissible limit. As the lower permissible limit is a function of the diluent partial pressure, the final selection of oxygen partial pressure will be deferred until the question of the diluent gas is resolved.

Furthermore, even though the problems of atelectasis, oxygen leakage, and combustion rate are qualitatively effected by oxygen partial pressure, as indicated above, quantitatively the effects are dependent upon whether or not a diluent gas is used and, if so, upon which gas is used and upon its partial pressure. Therefore, the quantitative effects will be evaluated in conjunction with the diluent gas analysis.

2.2.3 Diluent Gas Considerations

Reasons favoring the inclusion of a diluent gas are:

(a) The fan power required to transport a given quantity of heat to a heat exchanger is, for a given system, an inverse exponential function of the density. Because it is evident from the previous discussion on oxygen partial pressure that a relatively low pO_2 is desirable for the APOLLO vehicle, the inclusion of a diluent gas can appreciably reduce the heat exchanger fan power requirement. Figure I-2-3 shows the effect of adding both helium and nitrogen to the basic oxygen pressure of 187 mm (the minimum total pressure atmosphere that will maintain a normal alveolar pO_2 of 100 mm). Comparing two points on this curve it can be seen that by raising the total pressure to 360 mm, by adding 173 mm of nitrogen to the basic 187 mm of oxygen, the fan power can be reduced by a factor of four. For the APOLLO vehicle this means a reduction from 640 watts to 160 watts, or a difference of nearly one-half kw - a very substantial saving.

(b) For reasons previously given the presence of a diluent gas will probably decrease the hazard of atelectasis.

(c) It has been shown that, for a fixed pO_2, combustion time increases — and therefore fire hazard decreases — as the partial pressure of an inert diluent gas is increased. Figure I-2-4 indicates the effects of pO_2 and pN_2 on combustion time. The raw data used to prepare this curve were taken from a report by Simons and Archibald. These data were taken in the earth's gravitational field; consequently they are not quantitatively applicable to the zero-g condition where natural convection is absent. Quantitatively, the inhibiting action of the diluent gas upon flame propagation should be considerably enhanced in the zero-g field.

These considerations indicate that the diluent partial pressure should be as high as possible; however, other considerations place a limit upon this value. One consideration is the occurrence of aeroembolism in the event of rapid cabin depressurization; another is that the actual weight of the diluent gas in the cabin atmosphere, along with both the diluent leakage and purge weight, are linear functions of the diluent partial pressure; a third is that the weight of the vehicle pressure shell becomes a function of the atmospheric pressure when the pressure exceeds the permissible level for the minimum desirable skin thickness.

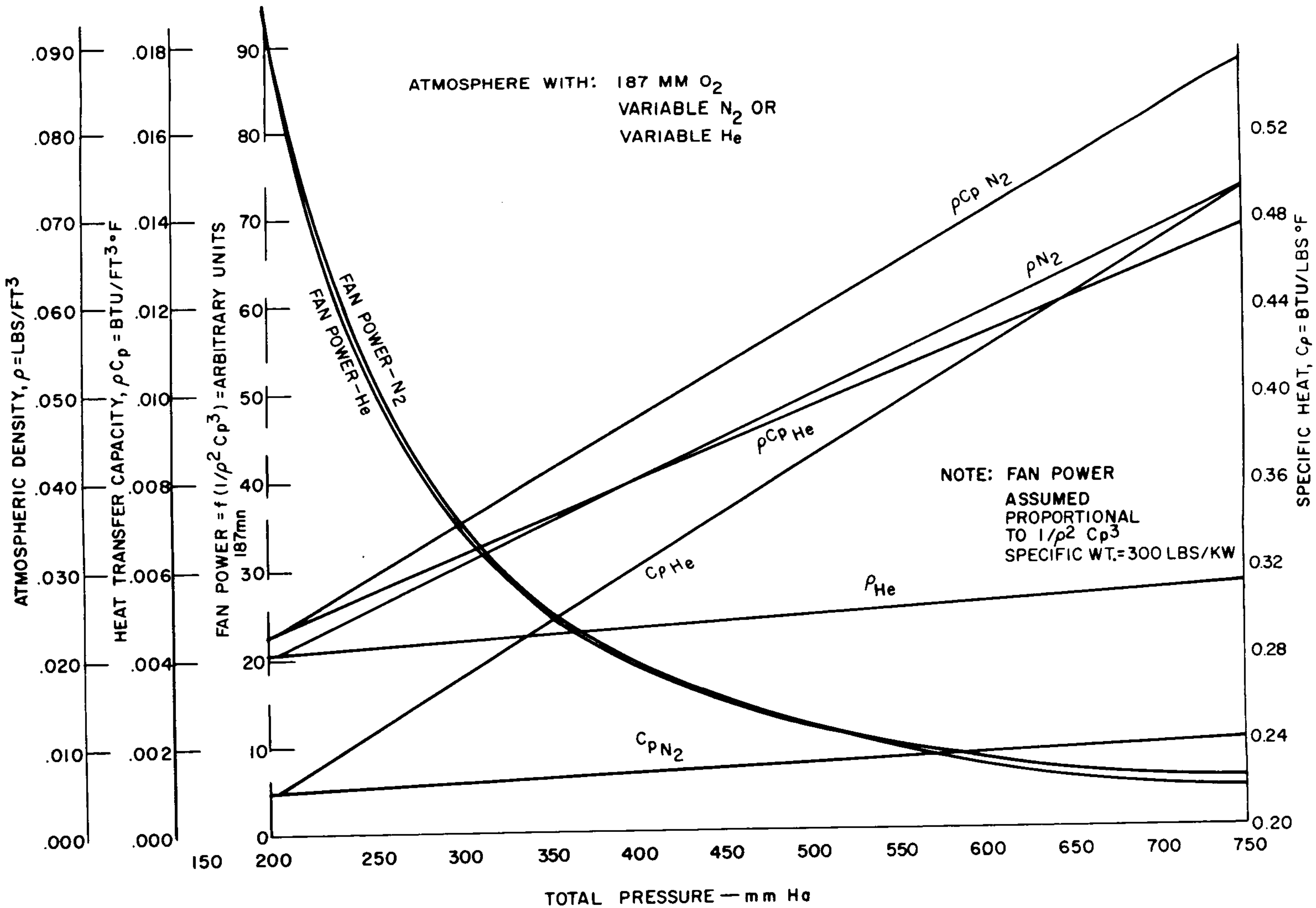

Figure I-2-3. Atmospheric properties and blower power as a function of atmospheric composition and pressure

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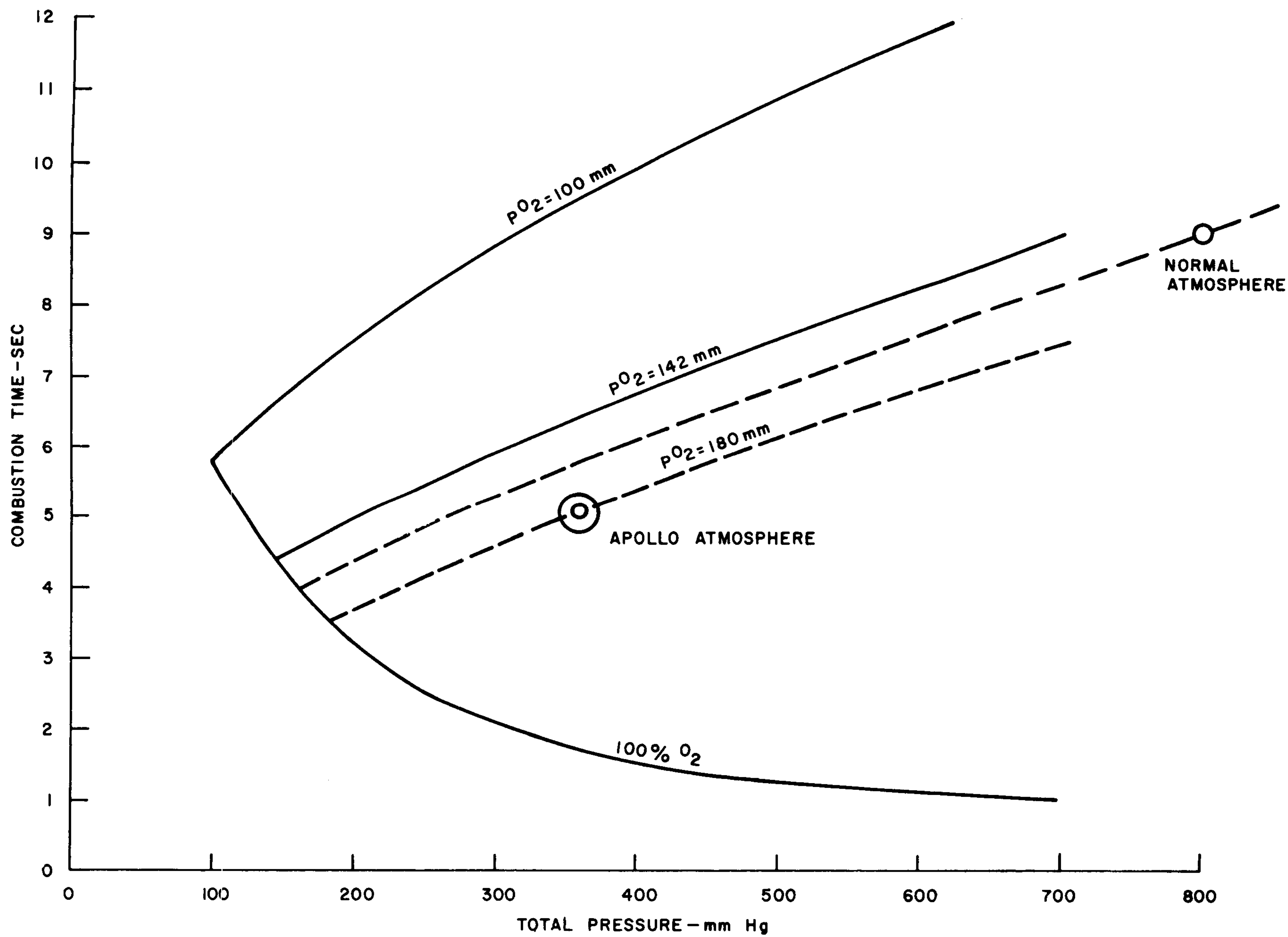

Figure I-2-4. Combustion time of paper strips as a function of pO_2 and total pressure (Derived from data taken from TR-59-36, data obtained in one-g field, diluent gas = nitrogen)

[redacted]

The quantitative weight assignable to leakage is a function of the gas used, as well as partial pressure, so this will be evaluated quantitatively when diluent gases are compared.

A disadvantage in using a diluent gas is that it will be necessary to regulate the rate at which oxygen is admitted to the cabin by means of a partial-pressure regulator rather than by means of a total-pressure regulator. This, however, is only a slight disadvantage, as will be shown later when the atmosphere sensing and electrical control subsystem are discussed.

Figure I-2-5 shows the weight versus atmospheric pressure relationship for all the applicable factors previously discussed. It is evident that the total system weight at first decreases and then increases as either nitrogen or helium are added to the basic 187 mm oxygen pressure, showing that the total system weight will be decreased by the addition of a diluent gas. The weight advantage obtained through using a diluent gas, along with the other reasons previously discussed, make obvious the desirability of including a diluent gas in the APOLLO vehicle atmosphere.

2.2.4 Diluent Gas Selection

Of the diluent gases suitable for use, the field can be narrowed rather quickly to nitrogen, helium, and neon. Practically no information is available on the long-term physiological effects of neon as a diluent gas. In addition, this gas required somewhat more fan power than either helium or nitrogen and is only slightly better than nitrogen from the aeroembolism point of view.

Consequently, for a detailed comparison only nitrogen and helium will be considered. The primary advantages of helium are that it requires somewhat less heat-exchanger fan power than nitrogen (see Figure I-2-3) and (assuming previous denitrogenation of the crew) permits a higher pressure-change ratio than nitrogen before the symptoms of aeroembolism develop. Also, if it becomes necessary to purge the cabin atmosphere, the stored weight of the helium required will be less than that of nitrogen.

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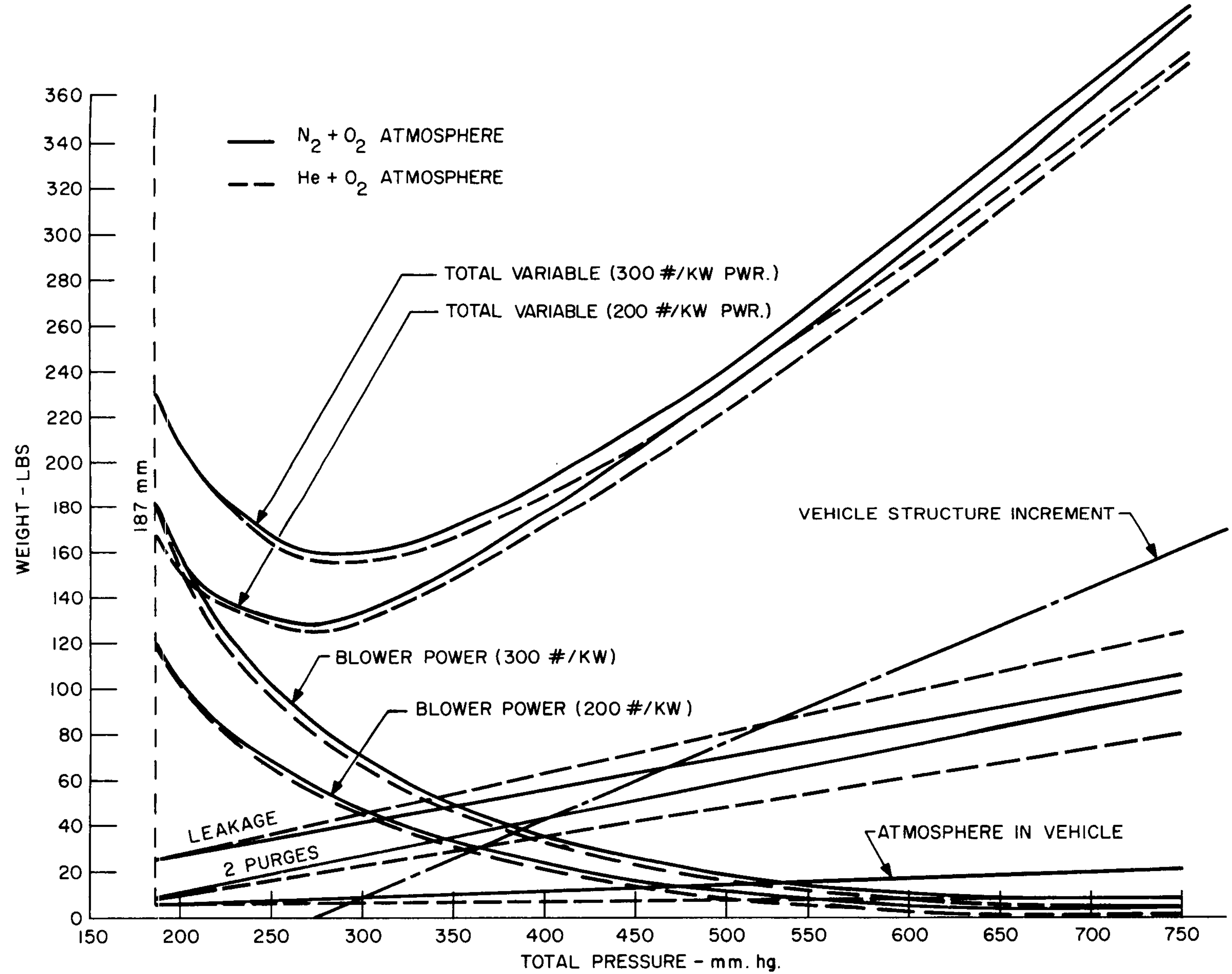

Figure I-2-5. Variable system weight versus total reserve for helium-oxygen and nitrogen-oxygen atmospheres

[redacted]

The disadvantages of helium are:

(a) For a given vehicle hole size, the weight of the system used to store the helium-oxygen atmosphere leaked is greater than that for the nitrogen-oxygen atmosphere. Figure I-2-6 shows the leakage weight comparison for the largest equivalent hole size expected in the APOLLO vehicle. This plot assumes that both the helium and nitrogen are stored as a high pressure gas in high strength titanium tanks.

(b) In the event of a puncture of the vehicle, it will, for the same initial pO_2 and total pressure, take less time with helium than with nitrogen as the diluent for the pO_2 to reach the lower critical value. As can be seen from Figure I-2-7, for an initial 360 mm total pressure, an initial pO_2 of 180 mm, and a 0.01 sq ft hole, it will take 63 seconds for the pO_2 to decrease to 90 mm with nitrogen and 50 seconds with helium. Consequently, with nitrogen as a diluent, the crew will have 26 percent more useful time to take whatever corrective action is indicated than they would with helium as a diluent.

(c) As can be seen from the data plotted on Figure I-2-8, which compares leakage weight in lb/hr versus hole size for helium-oxygen and nitrogen-oxygen mixtures, the leakage loss, through repairable-type holes, will be slightly greater for the helium-oxygen mixtures. The hole sizes contemplated on this plot are such that it may take some time to detect and repair them. By comparing curves 5 and 3 of Figure I-2-8, it can be seen that, for a hole diameter of 0.1 inch, the loaded weight of nitrogen leaked is 3.5 lb/hr and for helium 2.7 lb/hr. This, however, is not the complete story. Because the gas leaked is homogeneous and the volumetric leakage weight depends upon sonic velocity (which in turn depends upon the mixed gas properties), the amount of oxygen leaked with helium as the diluent is greater than when nitrogen is the diluent. This difference can be seen to equal (by comparing curves 1, 2, 4 and 6 of Figure I-2-8) 0.5 lb/hr more oxygen when helium is used than when nitrogen is used. This extra oxygen has a loaded weight of 0.9 lb. Consequently, the difference in leakage is 3.5 lb/hr versus 3.6 lb/hr; showing that, for a hole of this size, the loaded weight loss of the helium-oxygen mixtures is about 3 percent greater than it is for the nitrogen-oxygen mixture.

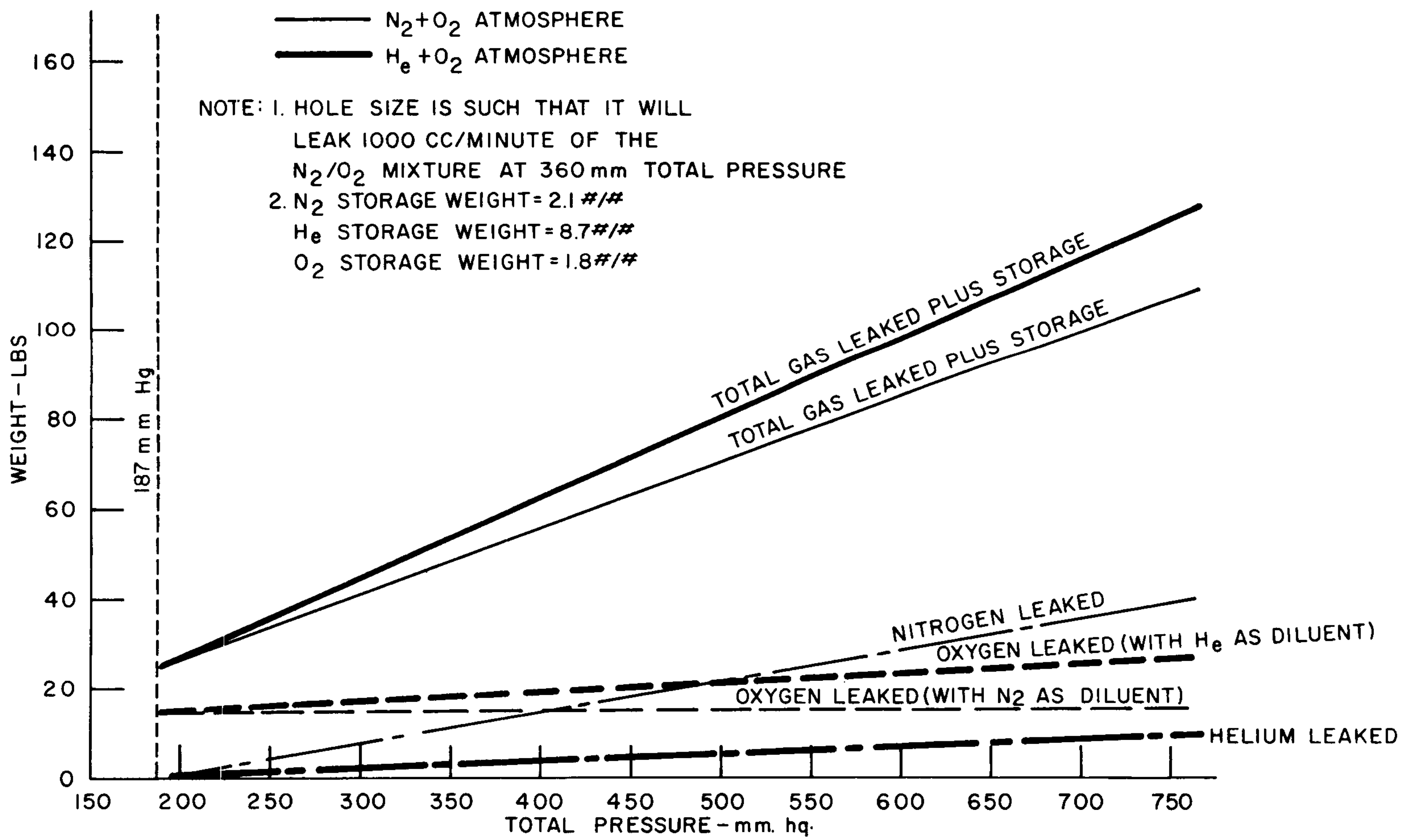

Figure I-2-6. Leakage weight penalty

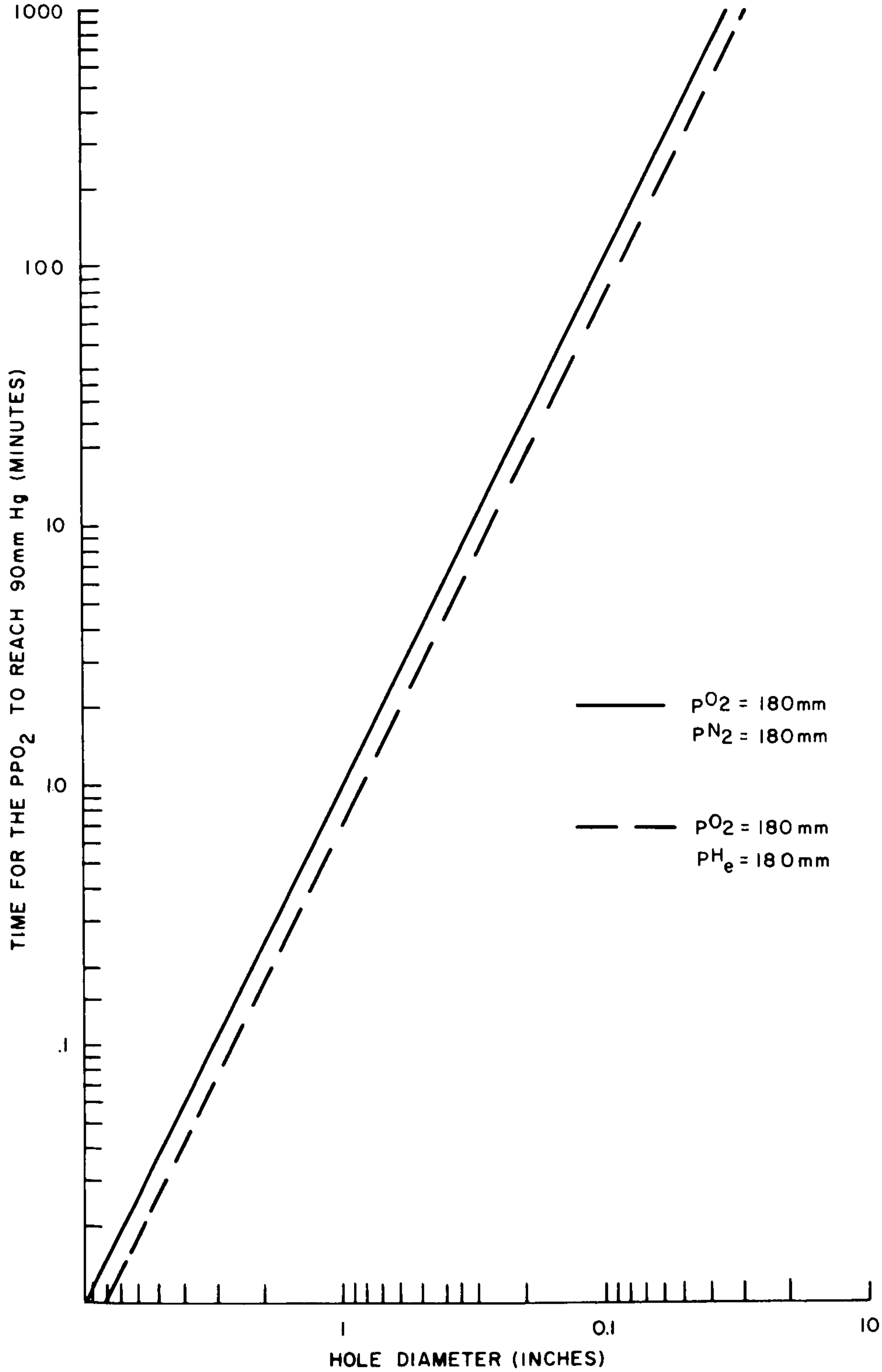

Figure I-2-7. Orifice size versus time (min.) for oxygen to reach the minimal partial pressure of 90 mm Hg considering both nitrogen and helium as the possible diluent gas at a total pressure of 7 psia and a free cabin volume of 300 cu ft

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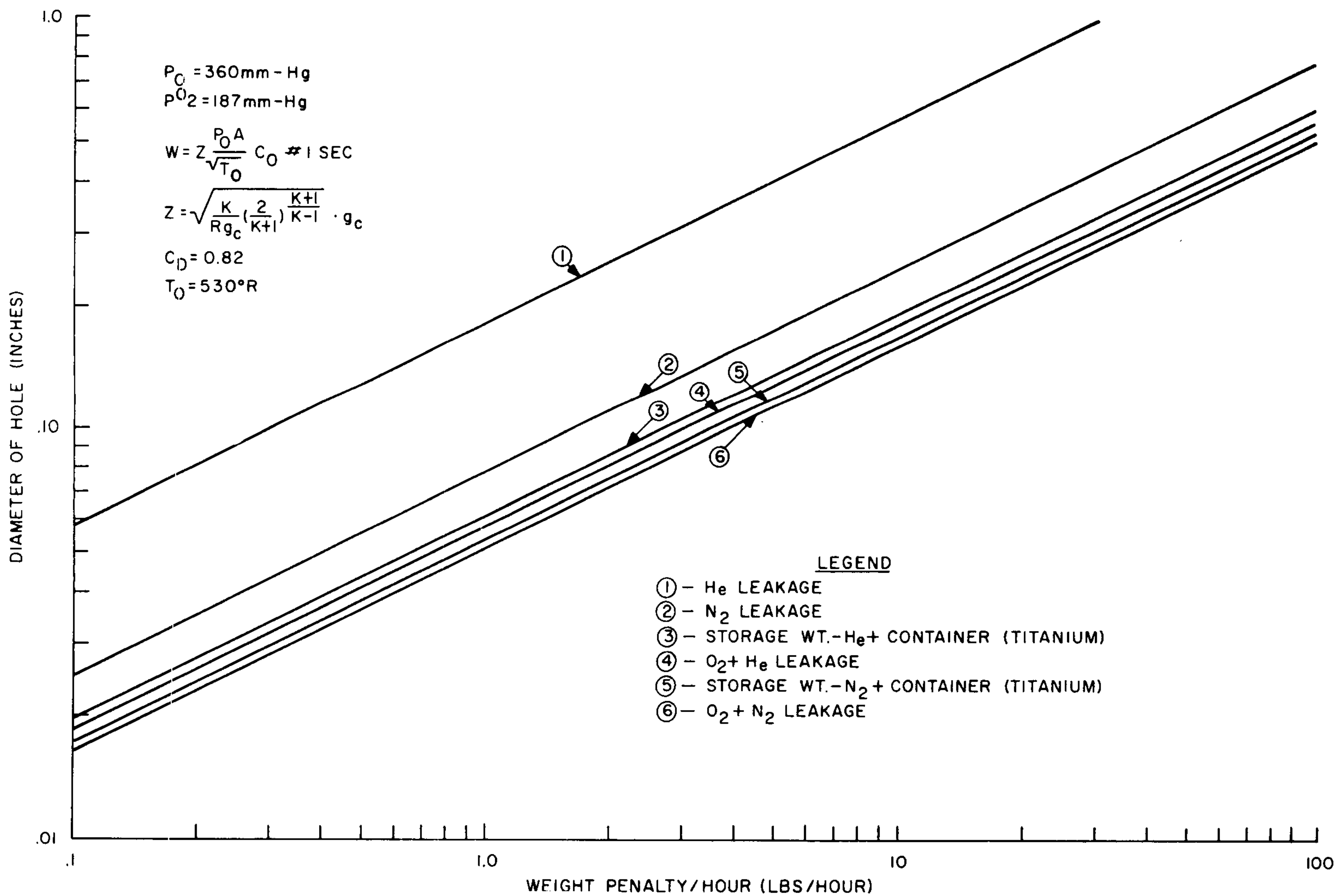

Figure I-2-8. Leakage weight as a function of hole size and atmosphere composition

[redacted]

(d) Some additional but hard to evaluate quantitatively, disadvantages of the use of helium are its adverse effect upon voice communication, its diffusion rate through the pressure vessel wall, and its expected inferiority to nitrogen as a combustion inhibitor in the zero-g environment. In the absence of natural convection and (assuming all blowers are turned off in the event of fire) forced convection, the only way that the oxygen molecules can reach the combustion zone is by diffusion through the gas cap surrounding the conflagration. Because of its lower molecular weight, helium will permit the oxygen molecules to diffuse more rapidly than nitrogen. Tending to offset this disadvantage of helium are its high specific heat and thermal conductivity, which will tend to cool the conflagration and, thereby (qualitatively at least) tend to decrease the combustion rate. However, the increased diffusion rate of oxygen in the helium is considered to be the more significant effect and, as stated previously, the net effect should be an increase in combustion rate with helium as the diluent.

Figure I-2-5 compares the system weight of all the variables previously discussed which are a function of atmospheric pressure for both nitrogen-oxygen and helium-oxygen atmospheres.

Several facts are evident:

(a) As noted earlier the total system weight for both diluent gases at first decreases and then increases with increasing total pressure.

(b) For each partial pressure of the diluent gas, the helium-oxygen system is somewhat lighter than the nitrogen-oxygen system.

(c) For a high total pressure atmosphere, where the primary advantage of using helium (to prevent aeroembolism in the event of rapid depressurization) is realized, the weight penalty is enormous.

Weighing all the factors presented it appears that the very small weight advantage — especially at atmospheric pressures near the optimum weight point-obtained by using helium as a diluent does not offset the disadvantages incurred through its use. Additionally, the present dearth of data on the physiological effects of both long term exposure to helium and lack of nitrogen confirms the decision to use nitrogen as a diluent gas.

2.2.5 Selection of Atmospheric Pressure

It is important that the atmospheric pressure selected for the APOLLO cabin be related to the operating pressure selected for the secondary pressurization system, because of the aeroembolism consideration mentioned earlier.

It is desirable to have as low a pressure as possible in the secondary pressurization system for several reasons:

(a) The mobility of an astronaut moving about an unpressurized cabin in a pressurized suit is inversely proportional to the suit pressure. Because it is anticipated that a certain degree of agility will be required to repair the pressure vessel, or perform almost any task which may be required, it is important that the secondary pressure be as low as possible.

(b) Atmospheric leakage from the suit, cocoons and the entire secondary pressurization and environmental control system - as well as their weight and structural design complexity - is a direct function of the secondary pressure chosen.

For these reasons a 187 mm, nitrogen free, atmosphere is chosen for the secondary pressurization system. If the pO_2 is maintained in the secondary system at 180 mm, the balance of the 187 mm required to maintain a normal alveolar pO_2 of 100 mm will be supplied by the pCO_2 and pH_2O.

The approximate pressure change ratio for which symptoms of aeroembolism develop in some individuals is 2.25. Consequently, if the pressure change ratio is held below this value, no problems should be experienced from aeroembolism. It can be inferred that if the pressure change ratio exceeds 3.0 aeroembolism will be a serious problem due to the amount of nitrogen that will be dissolved in the blood at the initial pressure of $3(187) = 561$ mm (assuming an initial pO_2 of about 180 mm).

It is interesting to note that for a given hole size, the greater the ratio of nitrogen to oxygen in the cabin atmosphere the greater will be the rate at which the atmospheric pressure will drop. Consequently, for a given initial pO_2, the greater the initial pN_2 the shorter will be the time for the pO_2 to reach a given value. As an example, for a given hole size the time required for the pO_2 to reach 90 mm will be 4.5 percent less for an atmosphere consisting initially of 187 mm of oxygen and 573 mm of nitrogen than it would be for an atmosphere consisting initially of only 187 mm of oxygen. This does not mean, however, that the time of useful conciousness will be decreased by the addition of the nitrogen gas. Due to the direct relationship, for a given inspired pO_2, between the alveolar pO_2 and the inspired diluent partial pressure, the nitrogen will actually increase, though not very significantly, the time of useful conciousness.

Figure I-2-9 shows system weight and relative fire hazard as a function of atmospheric pressure. It also shows the desirable range of atmospheric pressure, from which decompression to the secondary pressure of 187 mm will not present a problem from aeroembolism.

The decision as to what tradeoff is best is a difficult one. On the one hand it is desirable to minimize system weight and completely avoid any problem from aeroembolism by selecting a low atmospheric pressure; on the other hand the relatively intangible factors of increased danger from atelectasis and fire hazard tempt us to select a somewhat higher atmospheric pressure. Although the information presented in Figure I-2-9 indicates that a total pressure of between 265 and 275 mm Hg is optimum from a weight standpoint (depending on the weight penalty of the power supply); system considerations directed the selection of a total pressure of 360 mm Hg and a pO_2 of 180 mm Hg. The selected pO_2 of 180 mm is consistent with the conclusion already reached that it is desirable to hold the pO_2 close to the minimum level required to maintain a normal alveolar pO_2. 180 mm is also the partial pressure that was selected for the secondary pressurization system and consequently simplifies pO_2 regulation. Although the 360 mm system weight between 17 and 30 pounds more than the lightest system (depending on power supply weight) it still presents no aerombolism problem and provides somewhat more protection from atelectasis than would be achieved with the minimum weight system. In addition, it decreases the combustion rate below that of the minimum weight system by 13 percent.

[redacted]

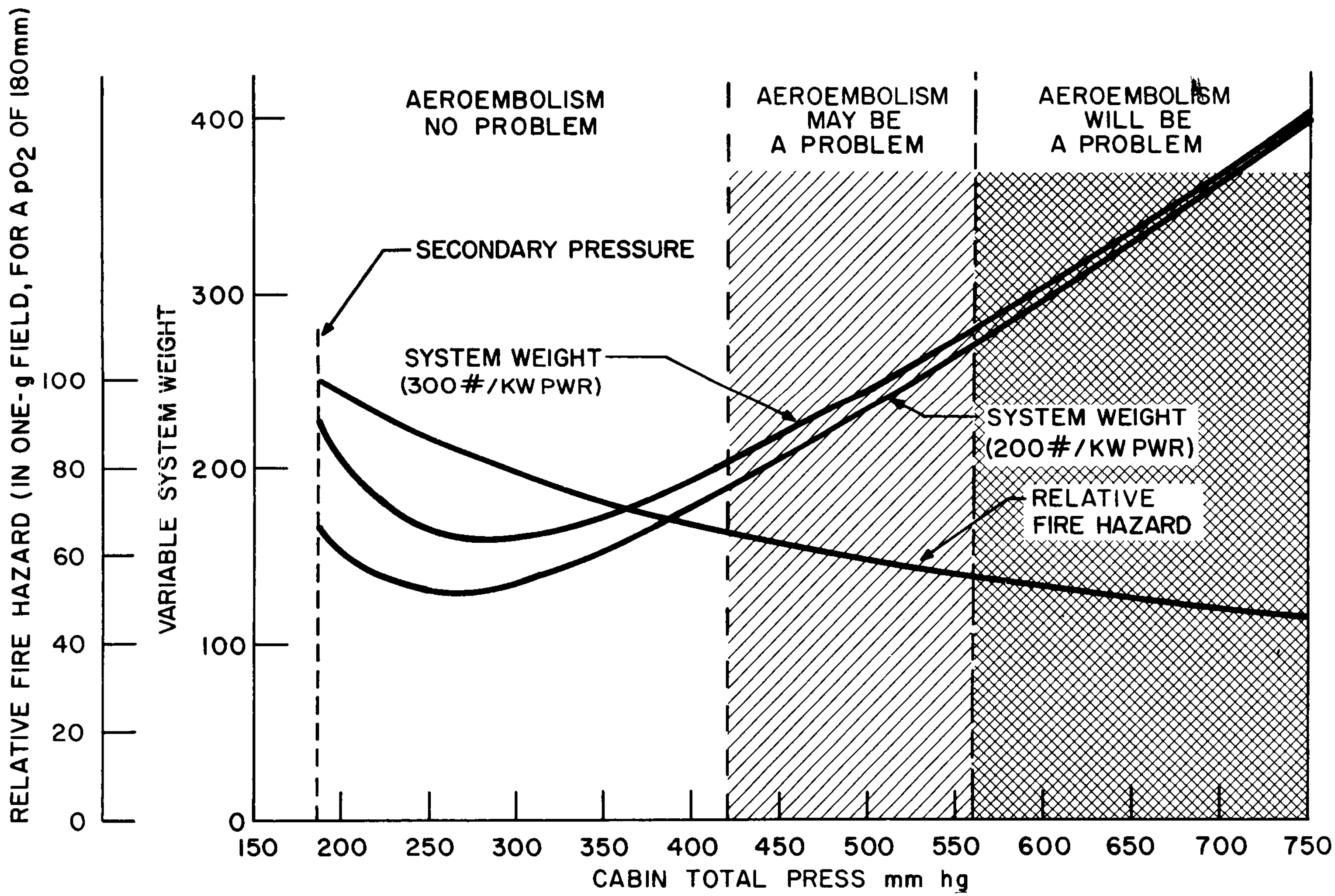

Figure I-2-9. Combustion time of paper strips as a function of p^{O2} and total pressure (derived from data taken from TR-59-36, data obtained in one-g field, dilaent gas = nitrogen)

[redacted]

2.2.6 Summing Up

Summing up, General Electric recommends a pO_2 between 170 and 190 mm and a total pressure between 350 and 370mm. This atmosphere satisfies the basic "ground rules" stated at the beginning of this discussion. In addition, a maximum pCO_2 of 8 mm and a pH_2O of 5 to 15 mm are specified.

2.3 MAINTENANCE OF THE CABIN ATMOSPHERE

The atmosphere control system is required to maintain the gaseous constituents, the ion balance, and the particulate matter concentration of the internal cabin atmosphere within predefined limits. Subsystems related to the environmental control system which are included in the following description are the fire detection and control system, the atmosphere sensing and electrical control system and the leak detection system. The thermal control system is discussed in Section 3.0.

The gaseous control system has the following primary functions:

(a) To store the oxygen required to replenish that consumed by metabolism and lost due to leakage — and possibly purging — and, to meter it to the cabin at a rate equal to the consumption rate, thereby maintaining the oxygen partial pressure within predetermined limits.

(b) To store the diluent gas required to replace that lost by leakage — and possibly purging — and to meter it to the cabin at a rate just sufficient to maintain the cabin total pressure within predefined limits.

(c) To remove water vapor, carbon dioxide, and miscellaneous noxious and toxic gases at a rate adequate to maintain their partial pressures within predefined limits.

2.3.1 System Description

The atmosphere control system is summarized and described with the help of Figure I-2-10.

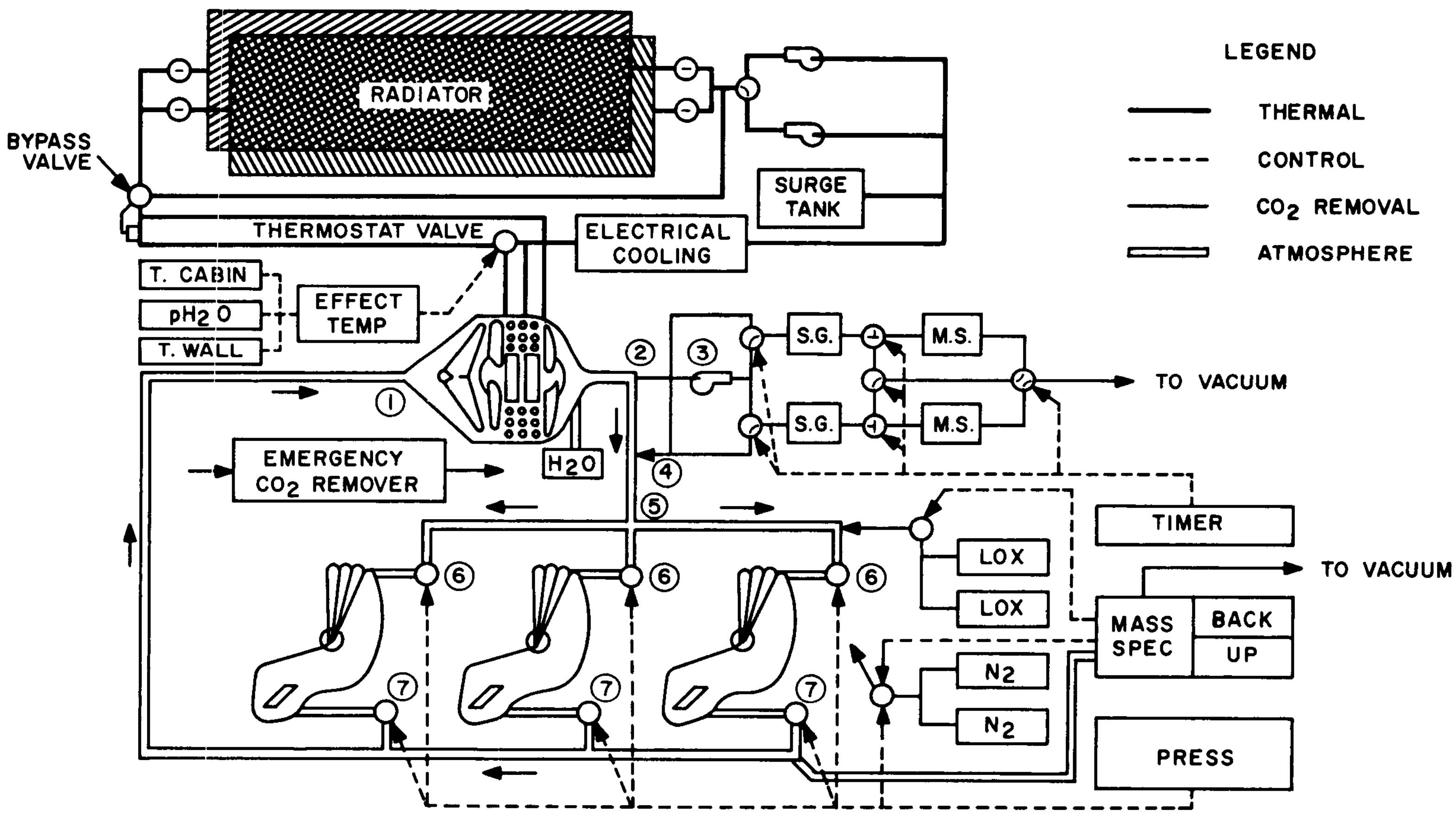

Figure I-2-10. Atmosphere control system schematic

2.3.1.1 ATMOSPHERIC CIRCULATION

The main atmosphere system circulates approximately 274 CFM. Starting the system description at point (1); the atmosphere enters the compartment heat exchanger where it is filtered, cooled, and dehumidified. At point (2) approximately 10 CFM are pumped from the main atmospheric stream, by the blower at point (3), through the carbon dioxide removal system and back into the main stream at point (4). At point (5) the flow is divided and one-third flows to each of the secondary pressure cocoons through the valves located at points (6). Under normal conditions the conditioned atmosphere flows from the discharge vents at the top of the cocoon, near the astronaut's head, into the cabin. The return flow is through intakes at the bottom of the cocoon near the astronaut's feet, through the valves located at points (7), and back to point (1). Should the cabin pressure drop below a preset limit, valves (6) and (7) close. They are automatically reopened when the canopy of the cocoon upon which they are located is closed. The atmosphere is then circulated through the pressure tight circuit thereby maintaining a shirt sleeve environment within the cocoon.

Each cocoon, additionally, contains an emergency conditioner which will supply oxygen and remove carbon dioxide for approximately 2 hours. This conditioner is portable and can be attached to a pressure suit. Its primary purpose is to provide time so that repairs can be accomplished.

2.3.1.2 CARBON DIOXIDE AND NOXIOUS AND TOXIC GAS REMOVAL

The atmosphere (approximately 10 CFM), which is drawn from the main atmosphere circuit at point (2), is forced by the blower at point (3) through the carbon dioxide removal system, shown schematically. After leaving the blower at point (3), the atmosphere (having a temperature of 45 F, a dew point of approximately 42 F and pCO_2 of less than 8 mm Hg) is passed through a silica gel canister where it is further dehumidified. (The adjacent silica gel canister returns moisture, thus being reactivated for the next cycle.) Passing through the activated charcoal and hopcalite, noxious gases are removed by adsorption and catalytic combustion, respectively. In passing through the molecular sieve, the carbon dioxide is removed by adsorption. Meanwhile, the other molecular sieve canister is being reactivated by exposure to space vacuum, as shown. When the core of each valve is rotated 90 degrees counterclockwise from the position shown, the operation of the system is reversed.

The silica gel and molecular sieve are contained within an air-to-air type heat exchanger so that the heat released during adsorption (approximately 1400 Btu per pound of water adsorbed and 400 Btu lb. of carbon dioxide adsorbed) can readily be transferred to the adjacent canister where desorption is occurring. The LiOH cannister shown between position (4) and (5) is a standby emergency system, containing enough chemical to remove one day's production of carbon dioxide. This system is intended for use in the event repairs should be required to the molecular sieve CO_2 control system.

2.3.1.3 GAS SUPPLIES

The oxygen is stored in the liquid form in four converters, each holding 28 pounds of oxygen. Three-fourths of the oxygen in each converter is the primary supply to provide for the maximum expected leakage and the average oxygen consumption of three 80th-percentile astronauts for 14 days. The remaining one-fourth of each converter contains a reserve supply for use in the event that one of the other converters fails to operate, the crew consumes more than expected, a purge of the cabin is required, or the leakage rate is greater than expected. The four converters are manifolded in parallel, with oxygen drawn from all four simultaneously.

The diluent gas, nitrogen, is stored as a high pressure gas in two containers. Each container holds 11 pounds. Oxygen and nitrogen will be admitted to the cabin by the sensing and control system as required to maintain their respective partial pressures in the vehicle cabin at 180 mm Hg.

2.3.1.4 ION BALANCE

It has not as yet been definitely established that ion control will be required. There is evidence to indicate that if the inspired air contains more positive ions than negative ions some adverse physiological and psychological effects occur. If it is determined that air ionization is required, it can be accomplished by addition of one or more tritium sources, each of which can produce on the order of a billion ions per second. Then positive ions are attracted to a negative electrode and neutralized, and the negative ions are discharged into the cabin.

2.3.1.5 PARTICULATE MATTER AND BACTERIA CONTROL

Particulate matter and airborne bacteria can both be removed by a particulate matter filter. A relatively high-efficiency filter is required to remove the 0.1 to 0.5 micron particle sizes which are, physiologically, the most dangerous. The filter, which is located in the air conditioning circuit just before the cooling coil, can remove both dust-borne and droplet nuclei bacteria with high efficiency.

2.3.1.6 CABIN ATMOSPHERE SENSING AND CONTROL

The atmospheric temperature, cabin wall temperature, pO_2, pN_2, pH_2O, pCO_2, total pressure, and the partial pressures of key toxic gases are sensed and the data displayed to the crew and telemetered to earth. Some of this information is fed back to the gaseous and thermal control systems to regulate their operation. Additional temperature, pressure, and flow rate sensors are placed at selected locations in the cabin atmosphere system to provide the information necessary to ascertain that the system is operating normally. These sensors may be interrogated by the APOLLO crew for system operating condition diagnosis.

A time-of-flight mass spectrometer is used to determine the gaseous composition of the atmosphere. In addition to providing the basic information required, the mass spectrometer is used to provide broad-spectra atmospheric data which can be telemetered back to earth for analysis by trained spectroscopists. Polarographic pO_2 and pCO_2 sensors are used as a back-up to the mass spectrometer.

2.3.1.7 FIRE DETECTION AND CONTROL

In order to prevent, as much as possible, the danger of fire aboard the APOLLO spacecraft, temperature sensors are applied to the electronic and other operating equipment. Equipment over-temperature will activate an annunciator to warn the crew of abnormal operations. A special manually operated fire extinguisher is located in the mission module, and another in the recovery vehicle. These extinguishers, charged with a non toxic, non conducting carbon dioxide activated foam, can be used to quench and extinguish visible and accessible hot spots or fires.

Emergency procedures will be established for use in the event of fire. These include such steps as: stop all blowers, de-energize all non-essential electrical equipment, stop oxygen supply, etc. The lack of convection will be a powerful fire-fighting aid. The oxygen molecules required to support the conflagration can, without convection, only reach the point of combustion by diffusion through the inert sphere of gas, consisting of the diluent and the products of combustion, surrounding the point of combustion.

2.3.1.8 LEAK DETECTION SYSTEM

The existence of small leaks can be determined by comparing present and past diluent consumption rates over a period of time of sufficient duration to cancel out the effect of changes in temperature, etc. Although providing an indication of the size of leak, no indication of location is available. An alternate means can provide both an indication of magnitude and general location: This General Electric Company concept uses ionization-type probes located about the exterior surface of the pressure hull. If a leak exists in the vicinity of the probe, the gas is ionized, thereby "activating" the particular probe. Since the probe location is known, the general area of the leak is thus revealed.

2.3.1.9 LOCATION OF ATMOSPHERE CONTROL EQUIPMENT

The existence of two modules requires that consideration be given to the location of the atmosphere control hardware both for normal operating conditions and in the event of the loss of the pressure integrity of either or both modules. For analysis, three cases are assumed:

(a) The normal case, where the vehicle successfully completes the mission, as planned, with both the command and the mission modules pressurized. The crew will all be in the command module for powered flight and re-entry. The balance of the time, one man will be in the mission module except, perhaps, for short periods of time when thrust is applied, either for course correction or for injection into, or ejection out of the lunar orbit.

(b) The mission module develops an irreparable leak just at the point of impractical abort return. This is defined as the point where it no longer pays, time-wise, to attempt an abort by applying retro-thrust. Beyond this point a circumlunar trajectory will provide the minimum return time to earth. For the purpose of this discussion, this point is assumed to be passed three days after lift-off. The return flight is assumed to take an additional half-day to the Moon and 3-1/2 days from the Moon to the earth, giving a total of four days without pressure in the mission module. This assumption of pressure loss at the point of impractical abort gives the longest emergency flight time possible.

(c) The command module irreparably loses its ability to hold pressure, just as the point of impractical abort return is passed.

For the first case, it is assumed that the hatch between the mission and command modules will remain open. Cooling, CO_2 removal, water vapor control, and oxygen replenishment can all be accomplished in the command module. Circulation between the modules is accomplished by directing the exhaust of a supplementary blower thru the hatch and into the mission module. A circulation rate of about 100 CFM is all that is required — and should easily be achieved — to maintain the temperature, CO_2, and humidity levels within the mission module at approximately the same value as in the command module.

Should case (b) occur, i.e. the mission module irreparably loses pressure, then all three men will remain in the command module. All of the environmental control equipment is located with them and will continue to operate to maintain a "shirt-sleeve" environment within the command module. Adequate gas reserves are carried to permit several visits — in a pressure suit — to the mission module, should this be required. Before the hatch is opened between modules, it is necessary to equalize the pressure within the modules. This will require the loss of something between one-half and full atmospheric content of the command module, depending on the size hole in the mission module.

Should case (c) occur, i.e., the command module irreparably loses its pressure integrity, then several choices are available. Either all three crew members can remain in the command module within their cocoons, or two can remain in the command module and one in the mission module, with the frequency of transfer back and forth depending upon how much of the gas reserves are used up (by module pressure equilibration and from leaky pressure suits) with each transferral and on the amount of gas reserves available. In addition to the question of gas reserves, other factors which will affect the decision as to whether or not a crew member will stay in the mission module are the habitability of the secondary pressurization system and the importance of the tasks which possibly can only be performed in the mission module. If the visitor to the mission module expects to stay for only a short time, it will not pay to pressurize the mission module while he is there. Instead, he can be supplied via an umbilical from the oxygen supply system to his pressure suit. If it is necessary or desirable for him to remain in the mission module for a long time, the oxygen required to pressurize the module can keep him for at least four hours without any additional supply. In addition, for this period of time, the CO_2 concentration will not build up to a dangerous level. Therefore, no additional atmospheric control equipment — except for the O_2 line required for pressurization — will be needed in the mission module. If it appears desirable to keep one man in the mission module for periods longer than this, he can use the portable lithium hydroxide emergency CO_2 control system, which has adequate capacity to remove one man's CO_2 for 3 days.

If, in addition to the problem of loss of pressure integrity of the command module, the pressure integrity of the environmental control system — which now is supplying conditioned atmosphere to the secondary pressurization system — is lost, then individual conditioners will be available as previously mentioned, within the cocoons.

In summary, all of the environmental control equipment plus a majority of the gas supply are located within the command module. Part of the gas supply, for optimum utilization of available volume, is located within the mission module. Small-diameter gas supply lines, plus the appropriate valving, will be available so that any or all of the gas supply can be discharged into either module as desired.

2.3.2 Atmosphere Supplies

Storage and supply systems for the oxygen and nitrogen to supply the APOLLO cabin atmosphere have been studied, compared and optimum system selected based on such factors as minimum weight and volume, complexity and maintainability, growth potential and fail-safe capability.

The cabin oxygen supply will be carried in four liquid oxygen converters of 28 lb capacity, each. Three fourths of the total stored oxygen in each converter is considered to be the primary supply, and one fourth is considered to be the reserve supply. The four LOX converters are manifolded in parallel, with appropriate check valving. Oxygen is drawn from the four converters simultaneously. This eliminates the loss of oxygen due to boiloff in a standby converter. The oxygen is admitted to the cabin air conditioning duct from whence it will pressurize the command module when the emergency pressurization cocoons are open, or the cocoons themselves if they are closed. A branch from the supply line conducts some of the make-up oxygen through a shut-off valve to the mission module, so that the mission module can be pressurized to 180 mm Hg. with oxygen if the command module is depressurized due to a leak. Alternately, the valve in this line can be closed if the pressure integrity of the mission module is lost.

The cabin nitrogen supply will be carried as a high pressure gas (3000 psi) in two containers. It will be supplied to the command module cabin through the appropriate pressure reduction and control valves. The nitrogen is not supplied to the cabin through the air conditioning circuit, since it is not used in the cocoons during emergency pressurization.

The control of the oxygen and nitrogen flow to the cabin is discussed separately. Descriptions of the selected systems and discussion of the merits of the system types that were considered are presented in this section.

The requirements for oxygen and nitrogen were derived from the metabolic requirements of the crew and a cabin leak rate of 1000 cc/min at cabin conditions (based on a conservative extrapolation of Project Mercury leak rates and the projected state of the art of cabin sealing). Sufficient additional supplies are carried to provide for purging of the cabin and to provide a safety factor on the above requirements. At this time the knowledge of the meteoroid population of cislunar space is scant, and the mechanics of meteoroid penetration of materials are still under study. The estimates of the danger of meteoroid penetrations, Volume VI, are necessarily, then, speculative. It is felt, however, that the APOLLO reserve supply is adequate for even the most conservative of the meteoroid impact estimates.

The possibility of replacing the diluent supply with a mixture of oxygen and nitrogen, stored together in a single container, has been considered. The oxygen - nitrogen mixture, of the same proportions that are desired in the cabin, can be stored either as a mixture of gases or as a mixture of liquids. If liquid storage is used, the metering device must be located so that it receives and controls the flow of the mixed liquid, rather than vapor from the storage container, since the nitrogen will vaporize at a lower temperature than the oxygen. The use of a mixed oxygen - nitrogen supply in place of the straight diluent supply offers an advantage in that it would simplify the partial pressure control system requirements for rapid repressurization after a purge.

The mixed storage system, however, commits a portion of the on-board oxygen to use in the cabin, and thus compromises the quantity available for use in the cocoons during emergency pressurization. A further disadvantage to the mixed storage of oxygen and nitrogen arises from the fact that this mixture will be supplied to the cabin to make up for the atmosphere lost due to leakage. While, for a given cabin pressure, flow losses to space through cracks or holes in the vehicle are proportional to the constituent gas concentrations in the cabin, the loss due to the diffusion of a constituent gas through spacecraft materials is proportional to the concentration divided by the square root of the molecular weight of the constituent. The ratio of oxygen to nitrogen in the stored mixture would then have to be modified according to the expected ratio of flow leakage to diffusion leakage. For these reasons, the more versatile, separate oxygen and diluent storage system is preferred for APOLLO.

2.3.2.1 OXYGEN SUPPLY

Presently, there exist in the industry several feasible methods of supplying the oxygen required for life support and pressurization aboard a space vehicle such as APOLLO. These are as follows:

1. Stored Oxygen
   a. in the gaseous state
   b. in the liquid state
   c. in the supercritical state (cryogenic gas at just above the critical pressure and temperature)

2. Chemical Reaction

   a. reaction of potassium superoxide (KO_2) with carbon dioxide (CO_2) and water (H_2O).

   b. decomposition of hydrogen peroxide (H_2O_2).

The stored oxygen supply systems were considered separately and as integrated with the space craft's on-board propulsion system. Regenerative methods of oxygen supply such as electrolysis of water, decomposition of carbon dioxide and controlled growth of algae were not found to be feasible because of the high weight and volume penalties which are characteristic of these systems.

A quantitative breakdown of the primary and spare storages is given in the following table.

	Weight of Oxygen	
Use	Primary	Spare
metabolic	1.6 (3) 14 = 67.2	0.2 (67.2) = 13.4
leakage	15.5	--
purge	--	12.6
Total	82.7 pounds	26.1 pounds

Grand total = 108.8 pounds

2.3.2.1.1 Integration of the Life Support and Propulsion Oxygen Supplies

Both the cabin oxygen system and the on-board propulsion system are essential to the safe return of the APOLLO crew to earth. The possibility of attaining greater reliability or lighter weight through the integration or interconnection of the two has been considered.

Concurrent studies of two APOLLO subsystems, i.e. the D-2 and R-3, have indicated that the 108 lb of oxygen required for the cabin and crew is in the order of only 2 percent of the oxygen required for on-board propulsion. The need for oxygen in the cabin will not, then, influence the selection of a bipropellant combination for propulsion, because the use of surplus cabin oxygen would be of negligible advantage to the propulsion system. Conversely, however, with the selection of an oxygen-using on-board propulsion system, then the desirability of integrating the cabin oxygen into the much larger propulsion supply must be compared with the desirability of using a separate cabin oxygen system.

Liquid oxygen for use as rocket propellant oxidizer, in accordance with MIL P-25508, will have a purity in excess of 99.5 percent USP. Federal specification BB-0-925 lists the same purity requirements for Aviator's Breathing Oxygen, with the additional requirements that the gas (or vaporized liquid) be odorless and free of contaminants and contain less than 0.02 mg. of water vapor per liter at STP. If propulsion system oxygen can be used advantageously in the cabin, it can be purchased to Aviator's breathing oxygen specifications or it can be dried in a molecular sieve cartridge and purified by filtration enroute to the cabin.

Of greatest relevance to the cabin system, is the oxygen that might ordinarily be vented overboard from the propulsion tanks to prevent overpressurization due to boiloff of the stored oxygen. This vented oxygen would be available for use in the cabin at a weight saving equal to its own weight plus the storage system weight for an equal amount of cabin oxygen, less the weight of equipment to meter it properly to the cabin. The current propulsion system studies indicate that through the use of efficient insulation and mounting techniques and scheduled midcourse guidance firings, the loss of oxygen due to boiloff can be avoided at the design use rate. If much less than the design quantity is required for midcourse guidance, some boiloff will occur and the tanks must be vented to relieve pressure.

The common storage of oxygen for the two systems is deemed inadvisable, since liquid oxygen can be stored in the cabin for the same tankage weight that would be required if it were stored with the propulsion supply. Relatively simple quantity gauging of a separate cabin oxygen supply provides an accurate means of monitoring the cabin oxygen use rate and time-to-go, whereas, in a combined system, the use rate would have to be obtained from an integrating flow rate meter, and the time-to-go would not be readily obtained from the quantity gauging on the larger propulsion storage tank.

However, a line from the propulsion oxygen storage system to the cabin oxygen system will allow the transfer of surplus propulsion oxygen to the cabin system in preference to jettisoning it overboard. In addition to providing an added margin of safety for the remainder of the mission, this connection makes possible the use of propulsion, oxygen in the cabin if emergency conditions in the cabin or cabin oxygen system make this necessary. This interconnection should preclude the possibility of the crew perishing from hypoxia while a surplus of oxygen is but a few feet away.

2.3.2.1.2 Cabin Oxygen Storage Systems

In considering the oxygen requirements for APOLLO, it was decided that a primary oxygen storage system and a spare storage system be provided. The primary storage is calculated to supply metabolic needs and to compensate for the oxygen leaked from the cabin. The spare supply is included to supplement the primary supply, in the event the crew requires more oxygen than is expected, the vehicle leaks more than is expected, or it becomes necessary to purge the cabin atmosphere to remove noxious or toxic gases.

Weight and volume trade offs were performed to compare high pressure, supercritical and liquid storage of oxygen, the reaction of potassium superoxide and the decomposition of hydrogen peroxide. Figures I-2-11 and I-2-12 compare primary oxygen storage weight and primary oxygen storage volume versus flight time for these 5 oxygen storage methods.

2.3.2.1.3 High Pressure Storage

Oxygen storage at high pressure has the advantage that it is capable of being dispensed, under zero g conditions, with no special hardware. Storage in three types of pressure

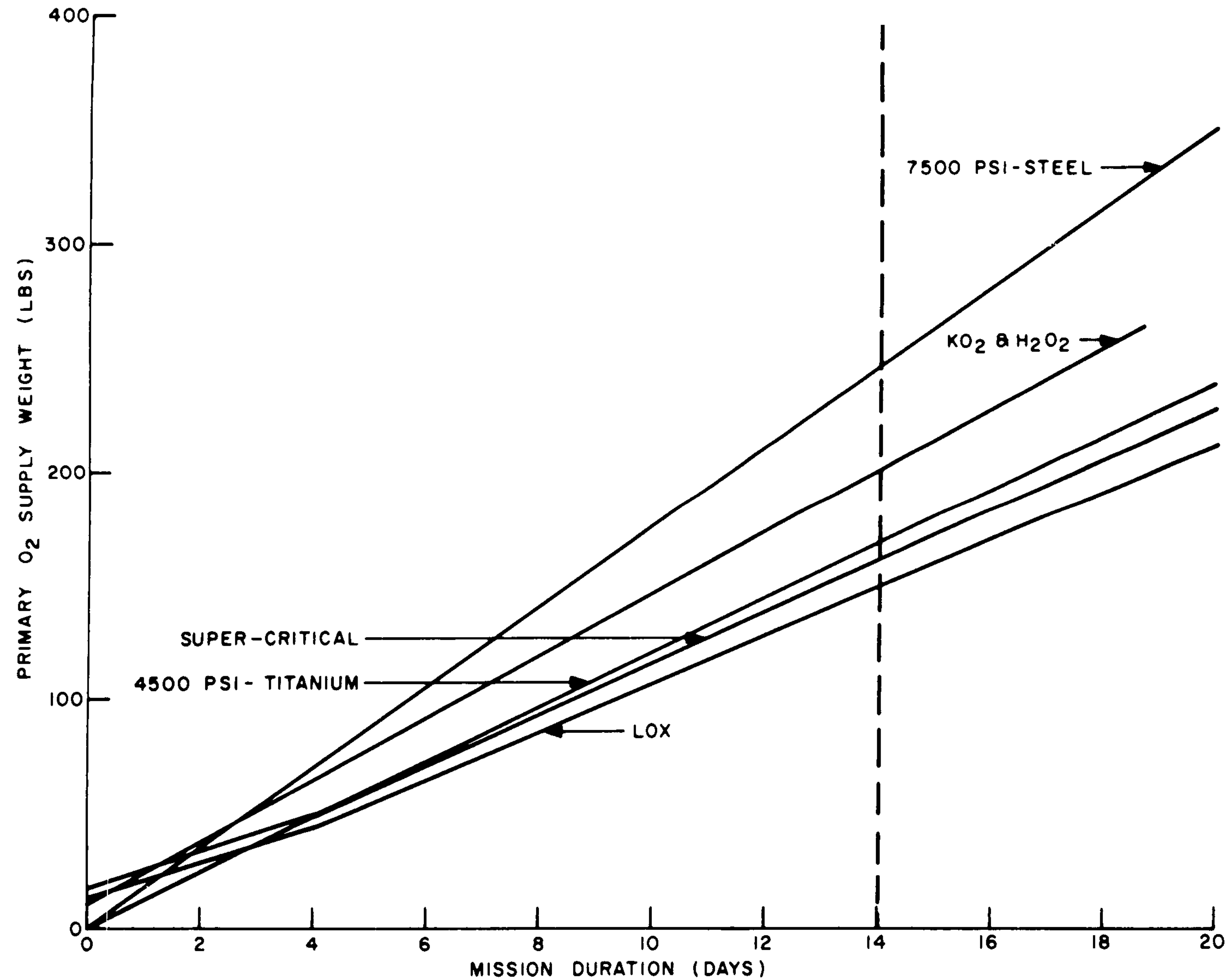

Figure I-2-11. Primary oxygen supply system weight versus mission duration

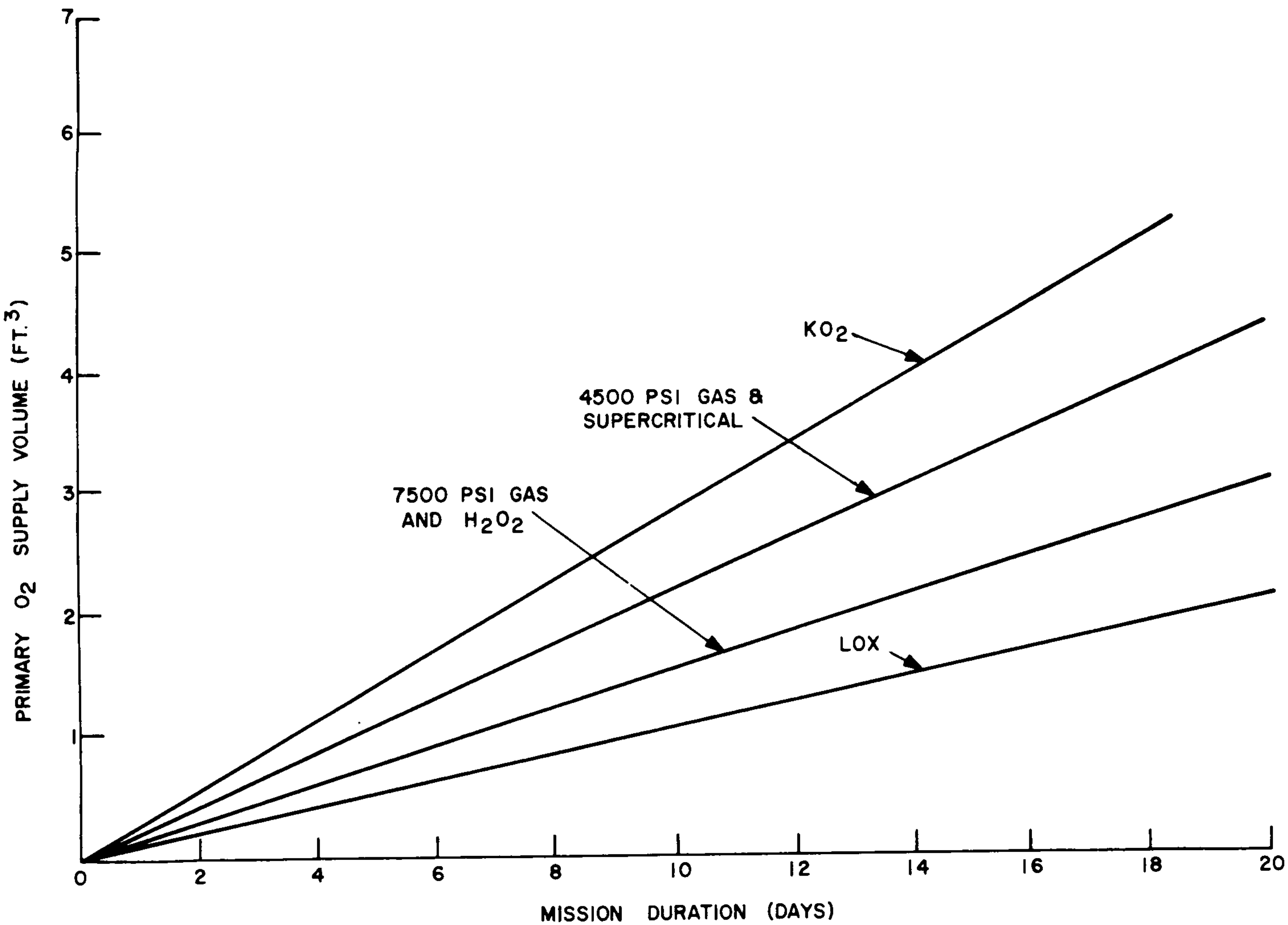

Figure I-2-12. Primary oxygen supply system volume versus mission duration vessels is compared; steel, filament-wound fiberglass, and titanium tanks. As can be seen from Figure I-2-13 a considerable spread is indicated for the fiberglass tanks. The data on fiberglass tanks that was used in preparing this estimate was received from several sources. The spread in the data, along with a history of trouble with fiberglass pressure vessels, indicates that the art of design and manufacture has not been perfected. However, this is not adequate reason for not considering fiberglass tanks, as advancements are being made in the state of the art.

When high pressure storage is considered, a trade off can be made between weight and volume. Up to about 300 atmospheres, the container weight for a given weight of oxygen is almost independent of pressure. However, above this pressure, the fairly rapid increase in the compressibility factor increases the weight of the container, since the weight is proportional to the compressibility factor. This is shown in Figure I-2-13 and Figure I-2-14 which compare weight, volume, and compressibility factor as a function of the container pressure.

2.3.2.1.4 Liquid Storage

The storage of oxygen as a liquid may be used to advantage in applications where the volume available for storage is at a premium. The disadvantage of the requirement for a cryogenic insulation system to maintain the oxygen in the liquid state must be offset by the advantages of reduced storage pressure and volume in these applications. For spacecraft application, where gravity is not available to separate liquid from gaseous oxygen in the storage container, an additional difficulty is encountered. This is the problem of designing a metering system that will maintain the required precise regulation of the mass flow of oxygen to the cabin with indifference to the state of the oxygen that it receives, or alternately, of designing a system that will assure the supply of gaseous (or liquid) oxygen to a conventional gaseous (or liquid) oxygen metering system. Although this problem has not been completely solved, it is being studied by several manufacturers, and there is reason to believe that a solution can be available for the APOLLO vehicle. The General Electric Company has also proposed to develop two zero gravity oxygen converter system designs in MSVD Proposals No. 070-844 and 071-002. The second design, which uses a honeycomb filler to

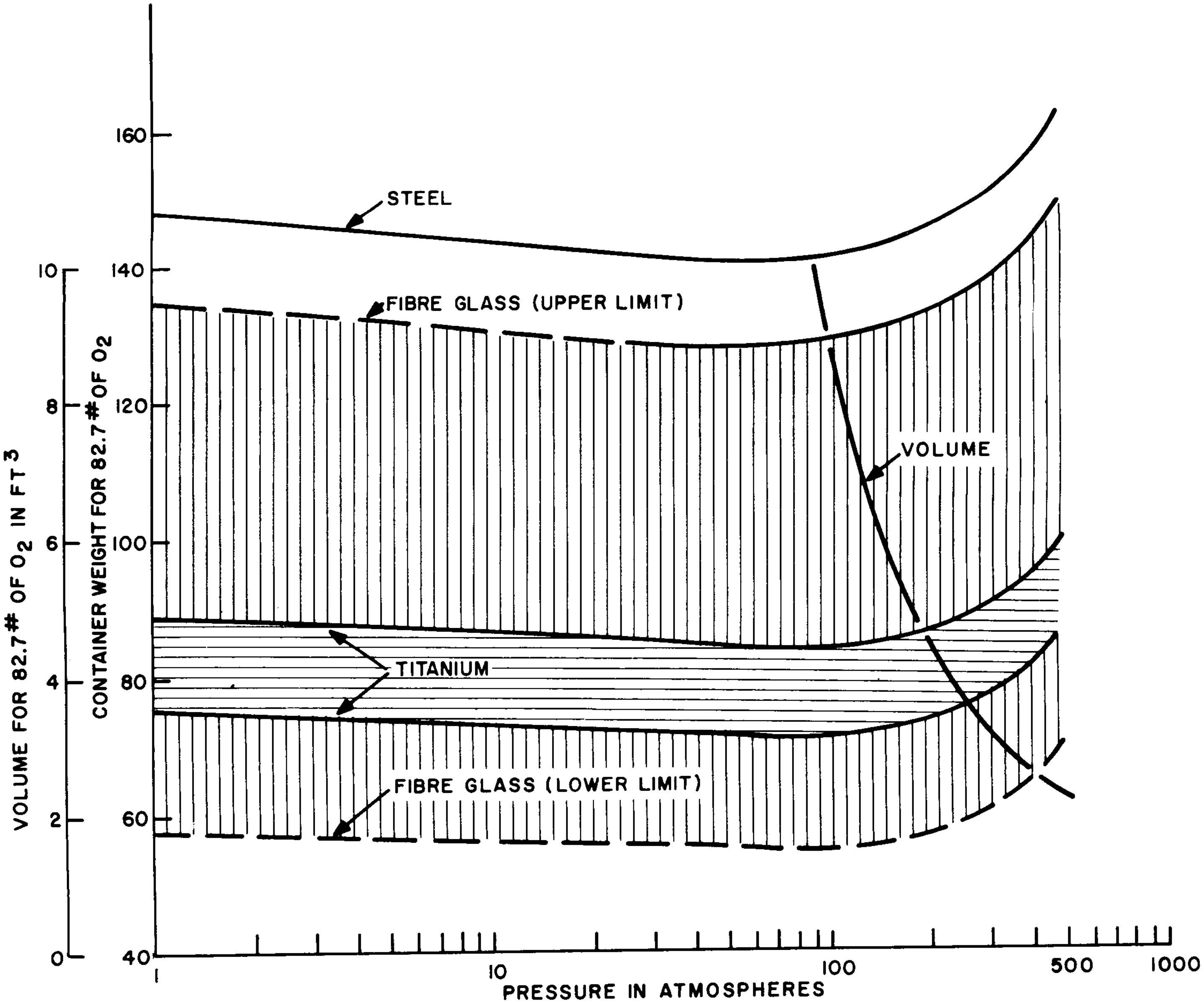

Figure I-2-13. Container weight and volume versus oxygen pressure at temperature (T) = 68 F

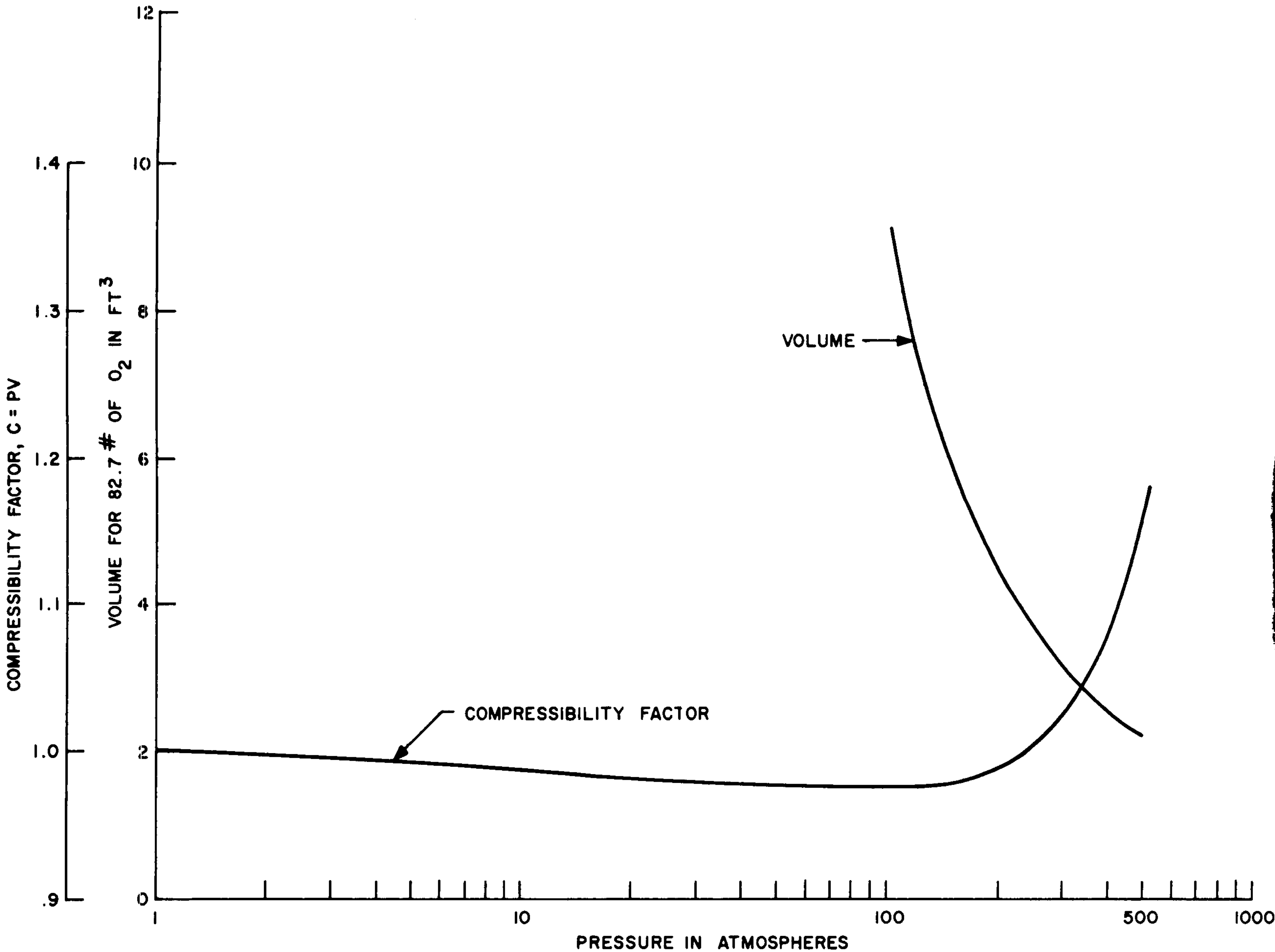

Figure I-2-14. Compressibility factor and volume versus oxygen pressure at temperature (T) = 68 F maintain liquid and vapor phase separation and to serve as the probe elements of a capacitance gaging system, is described in Appendix HF-A. Other converter concepts were also considered for possible further development. One assures that the oxygen being supplied from the storage reservoir to the metering device will be in the subcooled liquid state. Another uses a metering system that will regulate the desired oxygen flow to the cabin regardless of whether the oxygen that it receives from the storage reservoir is in the liquid or gaseous state or a mixture of the two.

The weight of the liquid oxygen storage system has been calculated at 0.8 pound per pound of oxygen, giving a total system weight, including the oxygen, of 1.8 pounds-per-pound. This weight applies to converters holding over approximately 25 pounds of oxygen. For smaller size converters the specific system weight will be slightly greater.

2.3.2.1.5 Hydrogen Peroxide Storage

Hydrogen peroxide, H_2O_2, will decompose when passed through a catalyst bed to form oxygen and water vapor ($2H_2O_2 \rightarrow 2H_2O + O_2$ + heat). By cooling the products of decomposition until the water is condensed, the oxygen is made available for use in the cabin atmosphere, and the water may be consumed by the crew. The amount of 90 percent solution hydrogen peroxide necessary to supply 82.7 lb of oxygen for the primary requirements during the mission would also provide 117 lb of water. The prime advantage of the hydrogen peroxide system is that the peroxide is stored at normal temperatures and pressures, resulting in a specific storage system weight of but 0.3 lb/lb of peroxide stored (which corresponds to 2.12 lb/lb of oxygen). Its disadvantages are that it requires a heat removal system to cool the products of decomposition and condense out the water vapor as well as a water separating device and sufficient valving and controls to regulate the process for safety under varying load conditions. The system must be designed for complete decomposition of the peroxide under any loading to assure pure drinking water. The weight of the catalyst, condenser, water collector, controls, and packaging has been estimated at 20 pounds.

2.3.2.1.6 Supercritical Storage

Supercritical storage of oxygen in this discussion means the cryogenic storage of oxygen just above its critical pressure. The container weight for supercritical storage will be about .96 lb/lb O_2. The storage of supercritical oxygen does not present the problems connected with liquid storage because the oxygen is still gaseous and thus no phase change or phase separation is needed. Zero g operation is, therefore, not a problem. Development is required, however, to regulate the heat leak into the Dewar to maintain the temperature and pressure above the saturated vapor phase line as the oxygen supply is used. Otherwise some oxygen would liquefy.

2.3.2.1.7 Potassium Superoxide Storage

The theoretical yield of the potassium superoxide system is 1 pound of oxygen for every 3.0 pounds of superoxide stored. KO_2 absorbs CO_2 in the process of liberating oxygen. Because of this, any fan power required to circulate the CO_2 laden cabin air through the superoxide bed is not charged against the oxygen supply, as this power is also required for this or any other method of CO_2 control. The volume of a superoxide system is assumed equal to the volume of required chemical.

2.3.2.1.8 Trade Off Factors

The system weights for the various methods of primary oxygen supply are given in Figure I-2-11. The volumes corresponding to the above systems are given in Figure I-2-12. A titanium tank, operating at 4500 psi, was selected for the comparison of gaseous supply because this represents the lightest proven method for gaseous storage (Refer to Figure I-2-13). A fiberglass tank may ultimately be lighter in weight, but this remains to be determined in development. A steel tank at 7500 psi is also compared. The 7500 psi pressure had been selected in a previous independent study (WADC TR 60-33). At this pressure the product of weight times volume reaches a minimum. Steel was chosen for its relative ease of fabrication and proven ability for use at this high pressure.

Examination of Figure I-2-11 indicates that the liquid oxygen converter affords the least weight penalty for oxygen storage in APOLLO. A supercritical oxygen system weighs only 13 pounds more. Volume of the supercritical system (Figure I-2-12) however, is more than 2 times the volume of an equivalent liquid oxygen system. Successful operation of the supercritical system is dependent primarily on balancing the heat leaked into the container against the heat required to maintain the oxygen above its critical pressure (50.1 atmospheres) and critical temperature (154.7 degrees K). A decrease in temperature or pressure will, in the early phase of usage, result in partial liquefaction of the oxygen. Excessive heat gain will result in too high a pressure in the container.

Operation of the liquid oxygen system, on the other hand, is not as sensitive to external ambient conditions. Passive heat transfer to the converter is not essential to the operation of the system, nor does it present a hazard. During normal operation enough heat will be transferred passively to the oxygen discharge coil to vaporize all the liquid that is required in the cabin.

A titanium pressure vessel for oxygen storage at 4500 psi weighs 10 lb more than the supercritical system and has the same volume. This 10 lb may, in fact, become nonexistent once a supercritical system is developed and the power required to regulate the supercritical pressure and temperature becomes known, i.e., the difference in weight at this time is not significant. However, the titanium pressure vessel weighs 23 lb more than the LOX system and occupies two times the volume, or 1.5 cu ft more. This additional weight and volume penalty is significant and rules in favor of the LOX system. The gaseous system remains, however, the simplest, and thus inherently the most reliable of any.

Overall systems considerations, including fabrication as well as those discussed above led to the choice of the LOX system for the APOLLO vehicle.

2.3.2.2 NITROGEN SUPPLY

In Section 2.0, Atmosphere Selection and Maintenance, nitrogen was selected as the diluent gas for the APOLLO cabin atmosphere. The supply of nitrogen can be stored as a gas at 3000 psi, as a cryogenic liquid, or at supercritical conditions for very nearly the same total system weight. The high pressure gaseous storage system has been selected as a result of this study because of its simplicity. It should be noted that the liquid storage system (including valves, heat exchanger, etc.) would occupy only half of the cabin volume that is taken by the 3000 psi gas system; however, this difference in volume is only 3/4 cu ft. The liquid storage system would be at a greater advantage if the total quantity of nitrogen required for the mission were greater. The fixed weight of control and vaporizing equipment for the liquid would then be more easily absorbed.

The amount of nitrogen required to be stored aboard the vehicle is the sum of the nitrogen required to make up that lost through leakage, plus the nitrogen required to repressurize the cabin after purging. It should be noted that a cabin purge is not a programmed part of the mission, but will arise from emergencies only, such as fire, or toxic gas control failure. The nitrogen leakage, from Figure I-2-6, is 12.5 lb for the total duration of 14 days, at the estimated maximum design leak rate of 1000 cc/min at cabin pressure. The amount of nitrogen required to repressurize the 300 cu ft cabin to a partial pressure of 167 mm Hg is 4.75 lbs. The total amount of nitrogen required, assuming two purges per mission, is then 12.5 + 9.5 = 22.0 lb. Comparisons are made below of the trade off factors of weight and volume associated with gas storage in a steel, fiberglass, or titanium container, cryogenic liquid storage, and cryogenic (supercritical) gas storage.

2.3.2.2.1 <u>Gaseous Storage</u>

Two factors are of importance in the gaseous storage of nitrogen; weight and volume. Figures I-2-15 and I-2-16 show the compressibility factor, volume, and container weight as a function of the nitrogen pressure of the container. As shown from a volume viewpoint, the operating pressure of the container should be as high as possible, since this results in the lowest volume. However, the container weight will be proportional to the compressibility factor. Referring to the Figures the container weight for a fixed amount of nitrogen will be constant with pressure up to about 100 atmospheres, above which the container weight will increase. It can be seen that the container weight increase is relatively small between 100 and 200 atmospheres while the volume is still decreasing rapidly. Above 200 atmospheres, the container weight increases much more rapidly while the corresponding decrease in volume is much lower. In the APOLLO vehicle, the decrease in volume obtained by increasing the pressure above 200 atmospheres

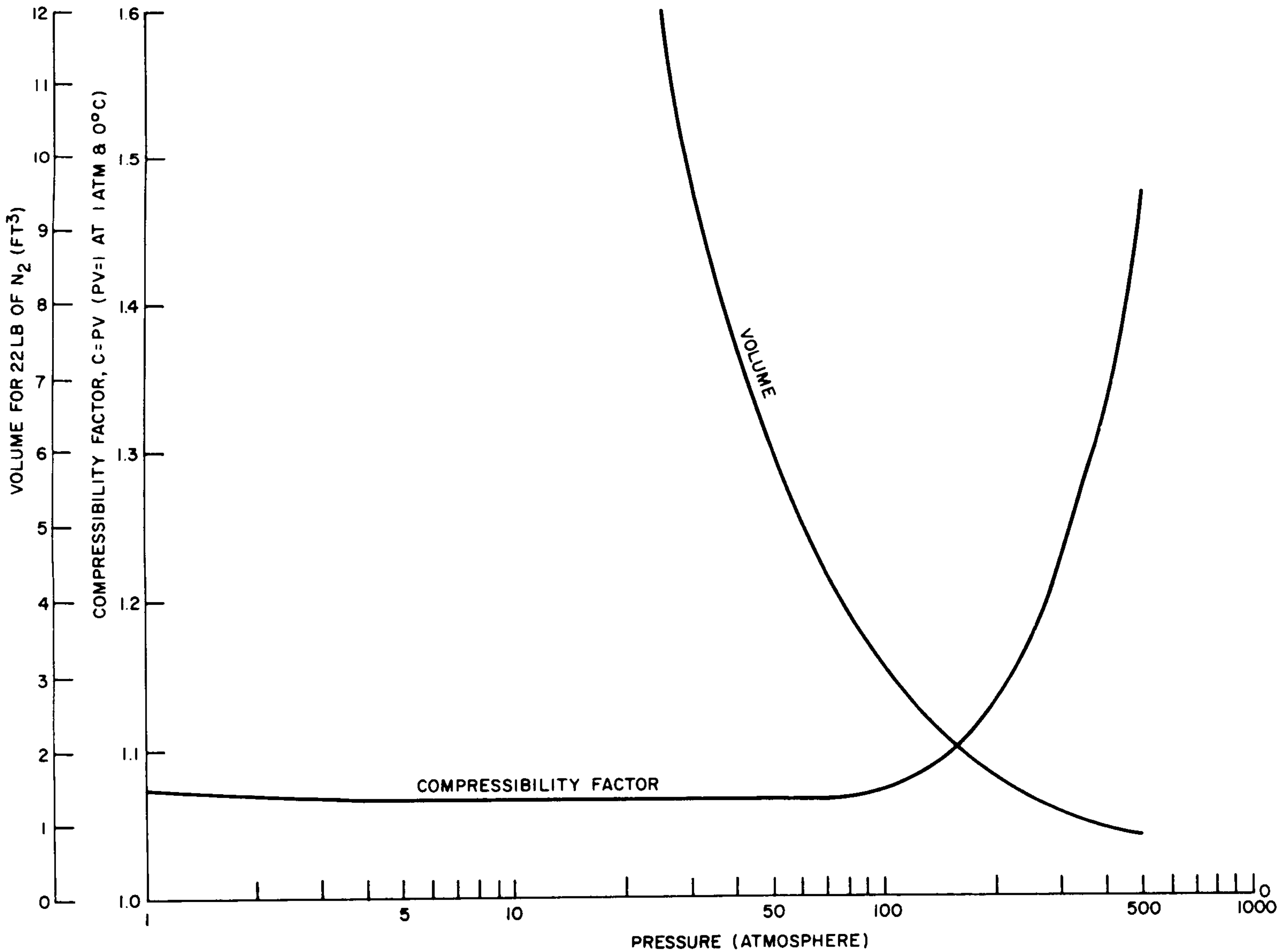

Figure I-2-15. Compressibility factor and volume versus nitrogen pressure (T = 68 F)

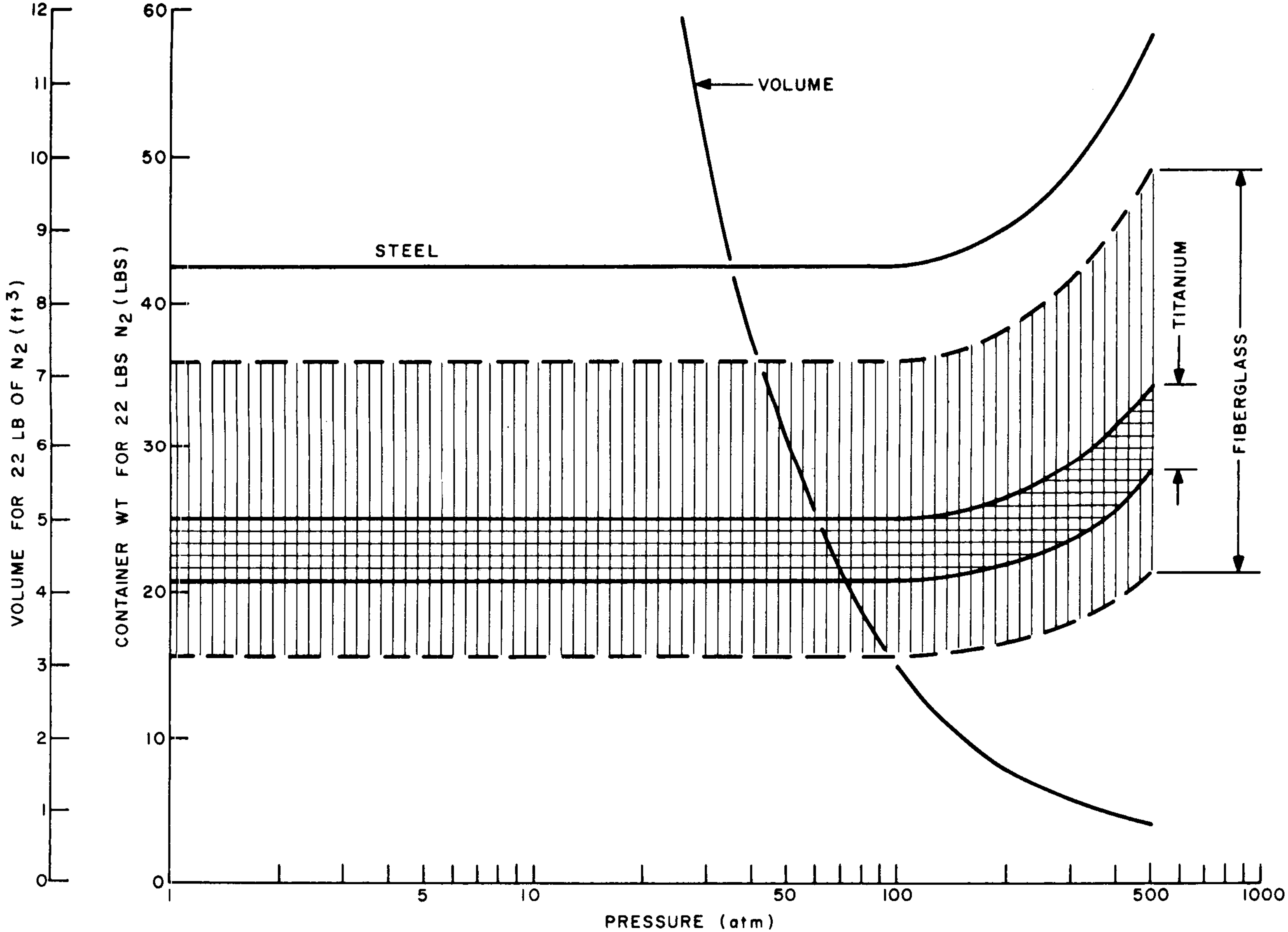

Figure I-2-16. Container weight and volume versus nitrogen pressure (T = 68 F) is not sufficient to warrant the additional weight penalty incurred. Therefore, the design pressure selected for gaseous nitrogen storage was 3000 psi (204 atmospheres).

Three materials were considered for pressure vessels; steel, titanium and fiberglass. Of these, steel has the highest weight to strength ratio and would consequently make the heaviest container. However, steel has certain advantages in relative ease of fabrication and familiarity of use. At 3000 psi, a steel container designed to store 22 lb of N_2 would weigh 45.3 lb. Thus, storage supply weight would be 45.3 + 22 = 67.3 lb. At this pressure, the storage supply volume would be 1.55 cu ft.

The weight factor assigned to the titanium tank is 2.2 lb of N_2 plus tank per lb of N_2 delivered. This number is based on existing designs, close to the design requirements for the APOLLO tank (i.e. operating pressure of 3000 psi, burst pressure over 6000 psi) which have been tested by the General Electric Company.

The material used in the fabrication of these tanks is a titanium alloy containing 6 percent aluminum and 4 percent vanadium. Heat treated, it has an ultimate tensile strength of 160,000 to 170,000 psi. Because of the difference in density between steel and titanium, this is equivalent to steel heat treated to 280,000 to 300,000 psi UTS. (The steel bottle used for the comparison above is fabricated of AMS 6434 heat treated to 180,000 to 200,000 psi UTS.) Both Airite Products, Inc. and Menasco Manufacturing Company, two leading manufacturers of titanium pressure vessels, make their tanks of this alloy.

An all-beta alloy, B-120VCA, developed by the Crucible Steel Company shows promise for pressure vessel applications. This alloy has an ultimate tensile strength exceeding 190,000 psi, and shows excellent promise of reducing the weight factor to 2.0.

In the case of the filament wound fiberglass pressure vessel, difficulty is encountered in assigning a weight factor (lb/lb N_2) for comparison. Data from several reputable sources indicate that the weight of container plus N_2 for a 5000 psi fiberglass vessel would be from 1.83 to 2.95 lb/lb N_2. A 3150 psi fiberglass tank was tested by the General Electric Company which had a weight of 2.3 lb/lb N_2. Although these tanks did not completely pass the testing to which they were subjected, the cause, and it is believed the cure, for the failure is known. The volume, as for the other tanks, is 1.55 cu ft.

A nitrogen supply system utilizing gaseous nitrogen at 3000 psi would be the simplest and, probably, the most reliable. No new techniques are required and zero-g operation is assured.

2.3.2.2.2 Liquid

The volume of liquid nitrogen required is $\frac{22 \text{ lb}}{49.4 \text{ lb/cu ft}} = 0.445$ cu ft at 172 R. The saturated pressure corresponding to this temperature is 82 psia. The liquid nitrogen converter (based on a similar size LOX converter) will weigh 1.1 lbs per lb of N_2. Therefore, the total storage weight would be (2.1) (22) = 46.2 lb.

The problems involved in designing a liquid nitrogen converter are greater than those involved in designing the gaseous storage system considered above. This stems from the conversion difficulties encountered in a zero-g environment, which does not provide the easy separation of the liquid and gas phases. The converter design problem has been discussed in the foregoing section on oxygen storage systems.

2.3.2.2.3 Supercritical

Supercritical storage of nitrogen in this discussion means the cyrogenic storage of gaseous nitrogen at low temperatures and moderate pressure, close to the critical point. The container weight for supercritical storage is calculated as follows:

Wt of dewar and mounting for 25 lb of liquid N_2 = 10 lb (based on corresponding elements of a LOX converter, good for 300 psia) $\therefore$ container wt. = $\frac{10}{25} = 0.4$ lb/lb. As the storage pressure is increased above 300 psia, the weight of the container must be increased proportionately. Now, volume is proportional to dia^3, surface area is proportional to dia^2 and, container weight is porportional to surface area. $\therefore$ container weight is proportional to vol$^{2/3}$.

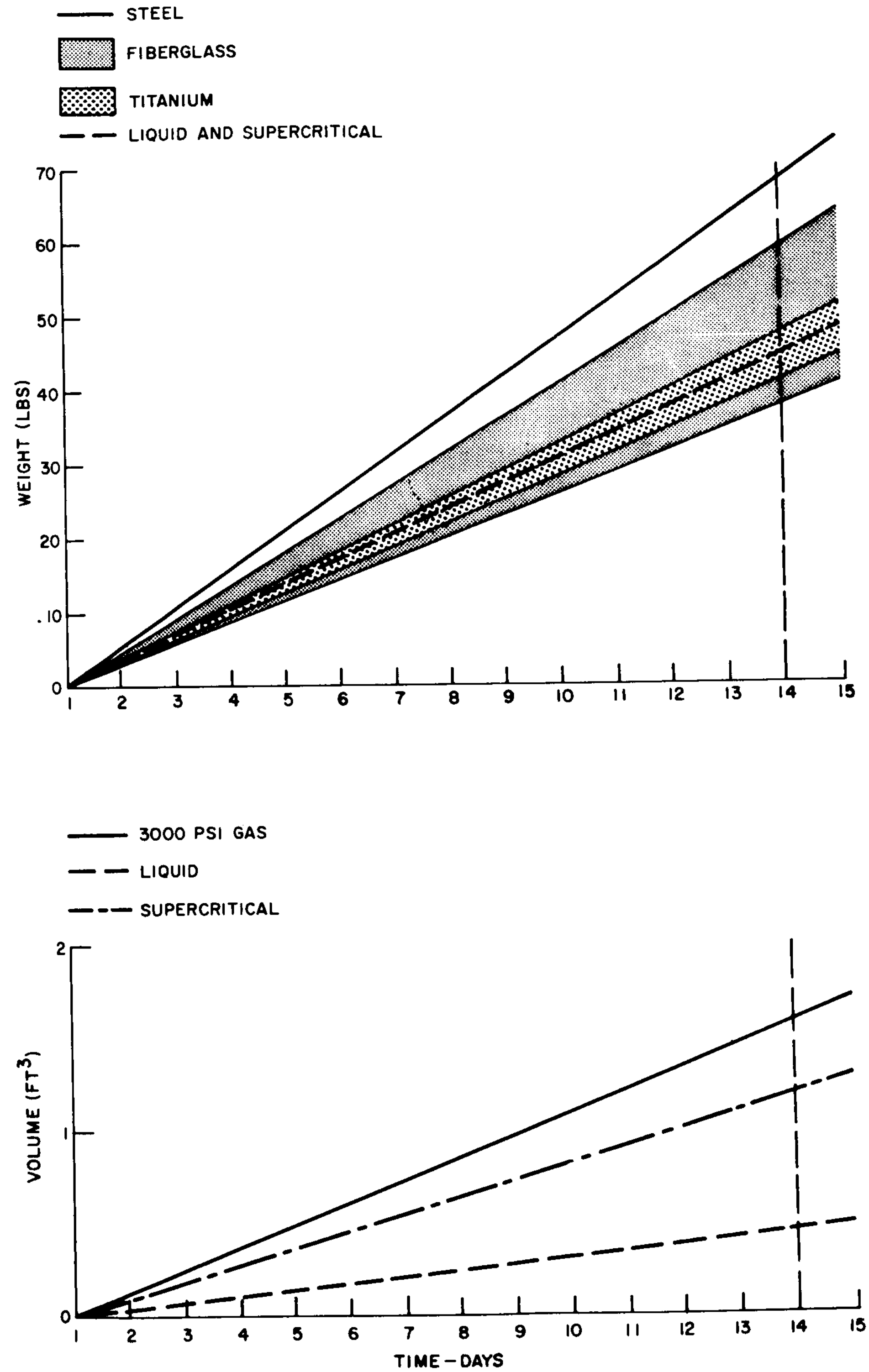

Figure I-2-17. Nitrogen storage versus time

The density of liquid N_2 = 0.69 gm/cc and the density of supercritical N_2 = 0.31 gm/cc (at 492 psia).

$$\text{Then container weight} = 0.4 \left(\frac{493}{300}\right) \left(\frac{.69}{.31}\right)^{2/3} = 1.13 \text{ lb/lb } N_2.$$

The volume occupied by the $N_2 = \left[\frac{22}{(0.31)\,(12.43)}\right]$ = 1.14 cu ft and the storage weight is (2.13) (22) = 46.8 lb.

The storage of supercritical nitrogen does not present the problems connected with liquid storage because the nitrogen is still gaseous and, thus, no phase-change or phase separation is needed. Zero-g operation is, therefore, not a problem. Development is required, however, to regulate the heat leak into the dewar to maintain the temperature and pressure above the critical point, thus preventing liquification.

2.3.2.2.4 Summary

High pressure storage of nitrogen has been selected for the APOLLO vehicle because of the simplicity of the system. With future developments anticipated in the state of the art of both titanium and filament wound fiberglass containers the weight should be no greater than that of a liquid supply system.

Figure I-2-17 compares the weight and volume of the various methods of storage as a function of mission duration.

2.3.3 Atmosphere Regeneration

2.3.3.1 CARBON DIOXIDE REMOVAL

It has already been determined that the partial pressure of CO_2 in the cabin atmosphere should not exceed 8 mm Hg. The problem now is to select the best method of accomplishing this.

An important question which must be resolved is whether or not it is advisable to recover the oxygen chemically bound in the carbon dioxide. A number of methods of disassociating carbon dioxide have been considered by General Electric, see Appendix HF-G.

The Fischer-Tropsch synthesis is the best known of these processes and could probably be developed into a flight-type system in time for the APOLLO vehicle.

If we assume complete recovery of the oxygen from the 75 pounds of CO_2 which is produced during the mission and a power weight penalty of 125 pounds per KW; the total system weight required to recover the 54.5 pounds of oxygen contained in the CO_2 is 120 pounds. This assumes that the Fischer-Tropsch synthesis will be used (with subsequent electrolysis of the water formed, and recovery of the hydrogen from the methane produced, in order to close the cycle). It is apparent that it is not advisable to attempt to recover the oxygen from the carbon dioxide as it would only require a 98.0 pound LOX system to store the same amount of oxygen. Therefore, this study will not place emphasis on carbon dioxide control systems which conserve the carbon dioxide removed from the atmosphere.

The following methods of controlling the carbon dioxide partial pressure are considered feasible:

(1) Freeze-out system: The carbon dioxide is frozen out on the surface of a heat exchanger, which is cooled by a heat pump.

(2) Potassium Superoxide: The atmosphere containing the CO_2 is passed through a KO_2 bed where the CO_2 is absorbed and oxygen is liberated.

(3) Lithium Hydroxide: The atmosphere containing the carbon dioxide is passed through, or over, a LiOH bed where the CO_2 is absorbed.

(4) Molecular Sieve: The atmosphere containing the CO_2, after having first been dried, is passed through a molecular sieve bed where the CO_2 is adsorbed. The molecular sieve is then reactivated by heat or exposure to vacuum.

An objective analytical study was made to compare the performance of these four methods. For an accurate comparison each system was modified to bring out its full potential. Figure I-2-18 shows the results of the study. In making these calculations it was assumed that the power weight penalty was 200 lb/kw for continuously operated equipment and 125 lb/kw for equipment which was not required to operate during launch,

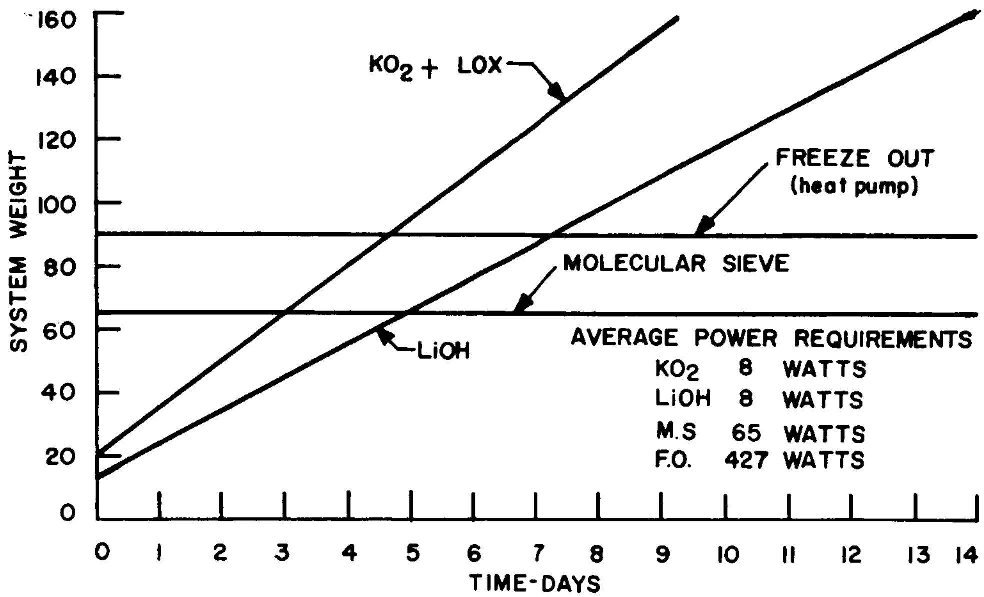

Figure I-2-18. CO_2 removal system comparison

re-entry or the dark portions of the lunar orbit. It is evident from Figure I-2-18 that the molecular sieve system is the lightest.

Figure I-2-19 shows a functional schematic of the molecular sieve system considered. The atmosphere containing CO_2 is forced through silica gel canister (1) where it is dehumidified, then through molecular sieve canister (1) where the carbon dioxide is adsorbed, back through silica gel canister (2) where the atmosphere is rehumidified, and then back to the cabin. Simultaneously, molecular sieve canister (2) is re-activated by exposure to space vacuum. Every half hour the core of each valve shown is rotated 90 degrees and the operation of the system is reversed to start the next cycle.

It is necessary to dehumidify the air before it contacts the molecular sieve because molecular sieves preferentially adsorb water to the exclusion of CO_2. With each cycle the silica gel becomes slightly more saturated with water because the dry air leaving the molecular sieve does not become completely rehumidified to its initial condition when passing through the second silica gel canister. Consequently, every fifth to tenth cycle the silica gel canisters must be heated to drive off the residual moisture.

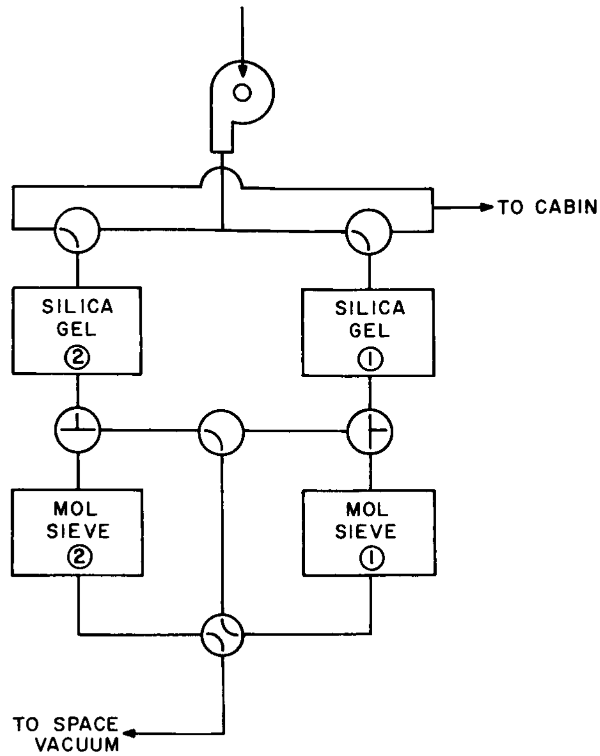

Figure I-2-19. Molecular sieve system schematic

An alternate method of removing the moisture from the atmosphere has also been investigated which involves freezing out the water. In this system, the water is frozen out on thermally regenerative heat exchangers, using, in one case, space and, in the other case, the latent heat of vaporization of the LOX as a heat sink. The heat sink is required to provide a small temperature differential between the circuits of the regenerative heat exchanger and to compensate for heat leaks into the system. Figures I-2-20 and I-2-21 show these systems schematically. The system shown in Figure I-2-20 uses the molecular sieve at approximately 35F to 40F, which some tests, performed on Linde 5A molecular sieves at General Electric indicated was the temperature at which the highest absorptive capacity for CO_2 was obtained. Subsequent data from Linde indicates that sieves 4A and 13X have high absorptive capacity at low temperatures. Therefore, the system shown in Figure I-2-21 was devised to take advantage of this characteristic.

The advantage of the freeze-out method of dehumidifying the atmosphere is that it eliminates the periodic power drain required to dry the silica gel. However, the intermittent power required to dy the silica gel will not impose a penalty on the power

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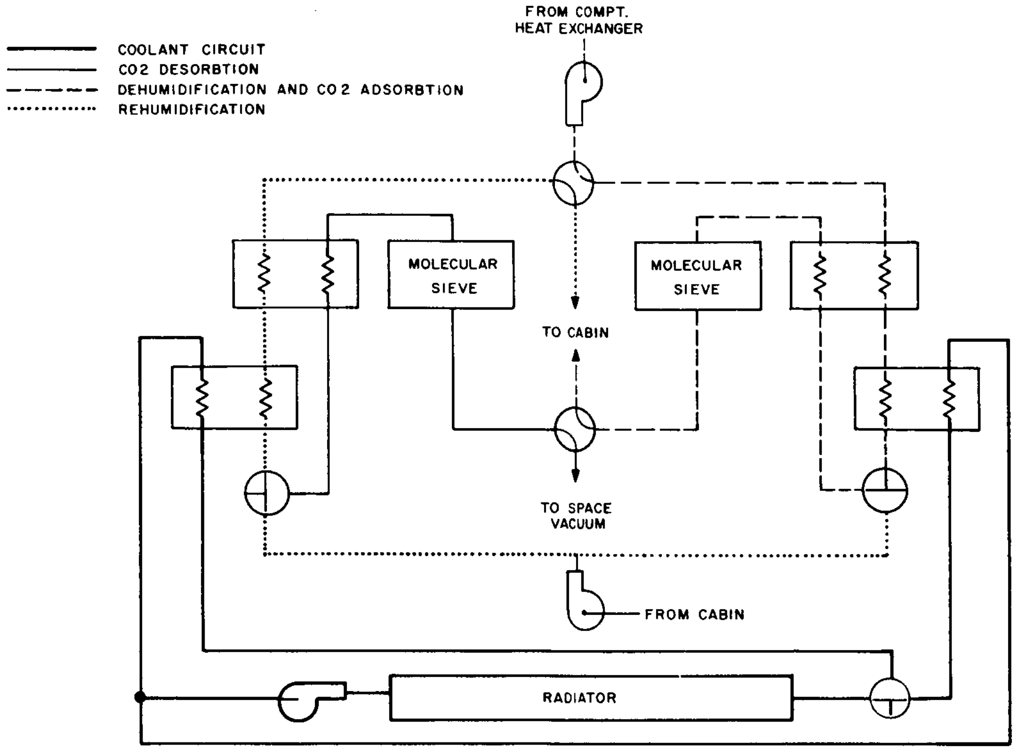

Figure I-2-20. Alternate molecular sieve system schematic

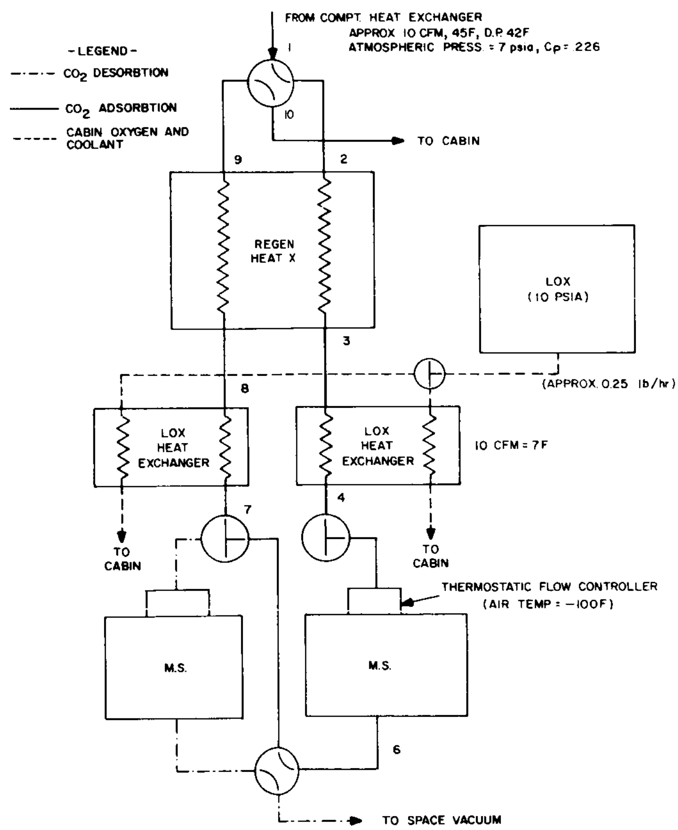

Figure I-2-21. Alternate molecular sieve system schematic supply system as presently conceived. The only requirement placed upon the silica gel is that the heater timing sequence be programmed during the lunar orbit phase of the mission so that the peak power demand does not occur during the dark portion of the orbit. In view of this, and the fact that the total heat sink available (through vaporization of and superheating from -305F to -100F, of 6 pounds of LOX per day) is only 34.4 Btu/hour, which is barely adequate, even with a high efficiency regenerative heat exchanger and a very well insulated system to dehumidify the air entering the molecular sieve, it is not considered advisable to attempt to use the LOX. In addition, a portion of the 34.4 Btu/hr will be required to provide a sink for the heat leaked into the LOX converters, so will be unavailable to the freeze-out system.

A system similar to that shown in Figure I-2-21, but which utilizes radiation to space instead of the LOX as a heat sink, is feasible. However, it would weigh about the same as the silica gel method and would be somewhat less reliable. Consequently, the silica gel method has been selected for the APOLLO vehicle. If, however, any substantial changes in the electrical power situation occur, which make it undesirable to draw the peak loads required by the silica gel, then the freeze-out dehumidification approach can be substituted.

Still another way of conserving electrical power would be to pump a hot fluid (in the order of 250F) through a heat exchanger located within the silica gel bed. The fluid could be heated by means of a heat exchanger incorporated within the skin on the sunlit side of the vehicle.

Although the system schematic shown in Figure I-2-19 could be constructed it would require a considerable amount of tubing and many valves. Figure I-2-22 shows a hardware concept which combines the whole system into an integrated package, with a single sliding valve which replaces all of the valves shown in Figure I-2-19. This valve is a vacuum tight multiple "O" ring sealed device operated by the diluent gas supply. If valve failure occurs, leakage of cabin atmosphere to space cannot occur due to a fail-closed feature. A manual override allows emergency use. Figure I-2-23 shows an operational schematic of the valve.

The large amount of finning in the beds causes inter-bed heat transfer of the heats of adsorption and desorption. This reduces the amount of electrical heat required for

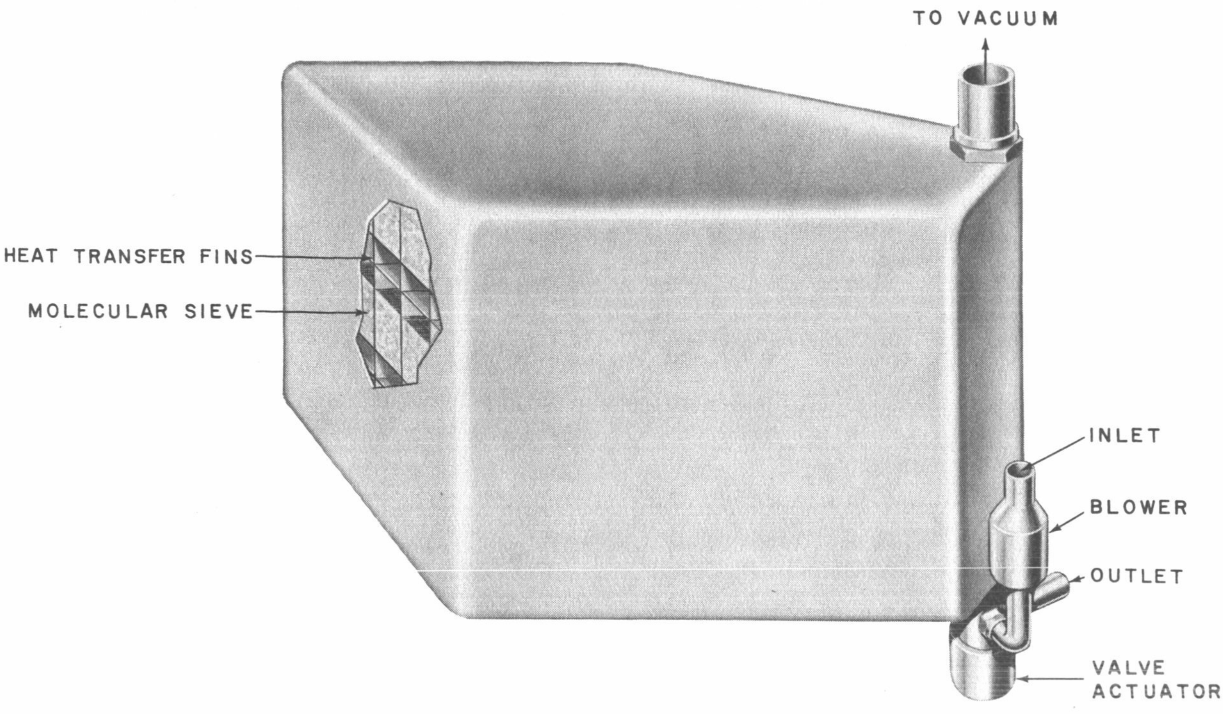

Figure I-2-22. CO_2 control system hardware concept

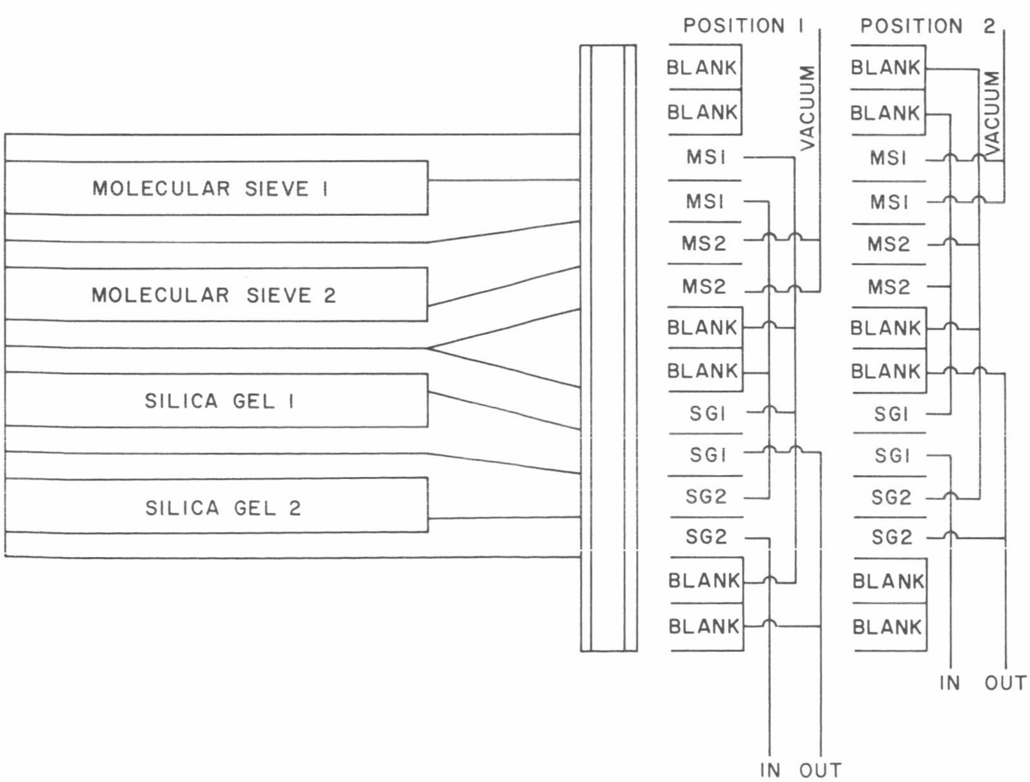

Figure I-2-23. Valve schematic

desorption of the water and increases the over-all efficiency of the system. During regeneration, the molecular sieve bed is exposed to vacuum from both sides. This results in a very short flow path for the CO_2 to reach the vacuum, thereby insuring a high degree of reactivation.

Figure I-2-24 shows an alternate hardware concept which, though not as compact as the system shown in Figure I-2-21, has the advantage that it is more easily repaired in the event that a pressure leak should develop. The separation of the canisters prevents the passive transfer of the heat of adsorption to the canister where desorption is occurring. However, coolant from the thermal control system can be pumped through the silica gel and molecular sieve canisters where the adsorption is occurring, and hot fluid from an external radiator (on the sunlit side of the vehicle) can be used to heat

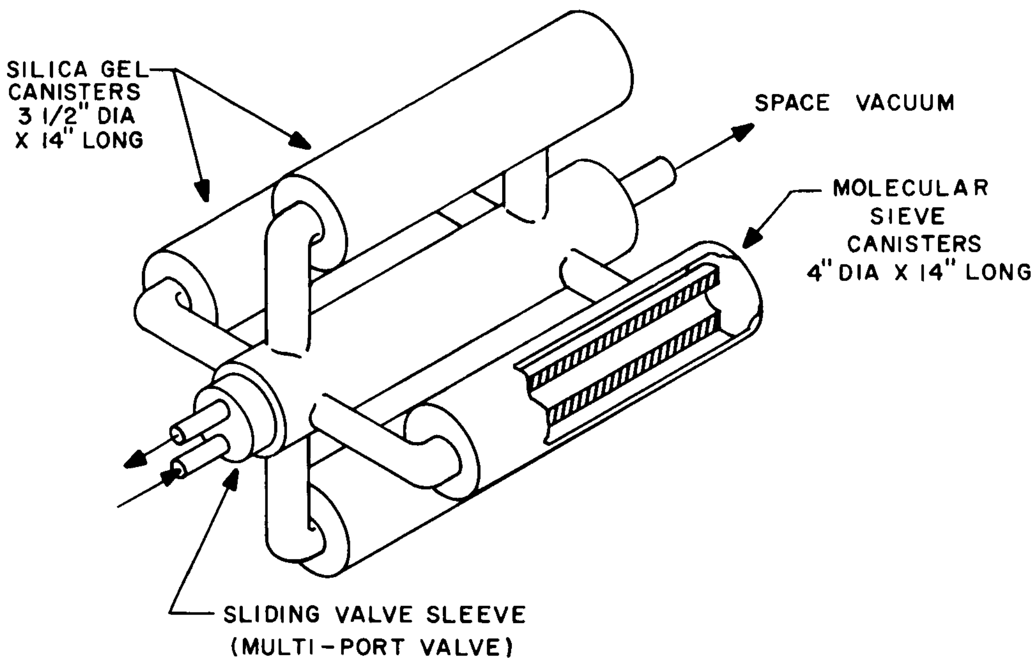

Figure I-2-24. CO_2 control system, hardware concept (alternate) the desorbing silica gel and molecular sieve. The high pressure nitrogen supply can be used in conjunction with an air cylinder to actuate the multi-port sliding valve. Since the air cylinder will vent into the cabin, the nitrogen used will not be lost.

Because it appeared early in the study that the molecular sieve system would be the lightest CO_2 control system for the APOLLO vehicle, the Hamilton Standard Corporation was asked to perform a parallel, but independent study. Applicable portions of the results of their study are included in this report as Appendix C. The overall weight and power requirements of the Hamilton Standard system are quite close to the requirements calculated by General Electric even though the systems are considerably different in detail.

Should the molecular sieve system fail, it may be possible, depending on the location of the failure, to operate the other half of the system while repairs are made. The pCO_2 level would rise under these conditions from the design value of 5 mm to about 10 or 11 mm. In the event that the failure occurs in a part of the system which precludes operation of one half of the system during repairs, then carbon dioxide control will be maintained by a portable emergency lithium hydoxide scrubber. The emergency scrubber, shown in Figure I-2-25, will remove three man-days production of carbon dioxide. The scrubber incorporates a novel method of CO_2 removal which was developed for the Discoverer bio-satellites. The main gas stream does not pass through the lithium hydroxide. Instead, the carbon dioxide diffuses from the main stream into the bed, thereby eliminating the pressure drop of the atmosphere though the lithium hydroxide and filter. In addition, as the atmosphere does not flow through the bed, and the hole sizes in the bed container can be micronically sized, there is no contamination of the atmosphere with the highly irritating lithium hydroxide dust.

The recommended molecular sieve approach to CO_2 control has been demonstrated to be feasible by several investigators, e.g. General Electric and Hamiliton Standard. Design of a lightweight, reliable and maintainable system does not appear to pose a serious problem. Reasonable mission growth can easily be handled by this type system even if requirement for collection of CO_2 (for subsequent processing) is imposed. CO_2 freeze-out is another approach wherein the CO_2 may be easily diverted to a subsequent

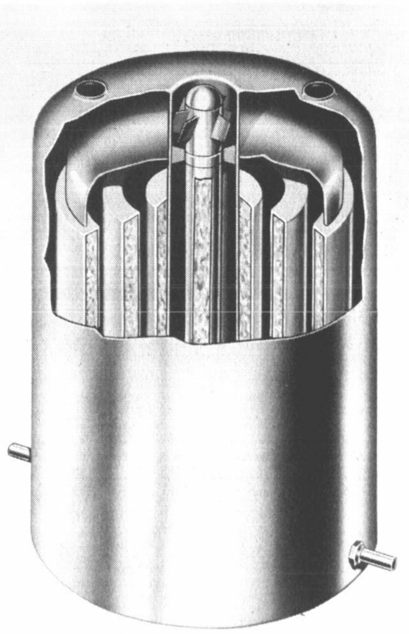

Figure I-2-25. Emergency LiOH carbon dioxide remover

processing system. Recent analysis of a novel General Electric CO_2 freeze-out concept using direct radiation to space for heat rejection indicates substantial saving in system weight as compared to the heat pump approach. This novel concept is reviewed in detail in Appendix A.

2.3.3.2 HUMIDITY CONTROL

The humidity of the cabin atmosphere will be kept between 5 and 15 mm Hg partial pressure. (See Appendix D.) Since it will be initially at or near this level, the H_2O removed must be equal to the moisture added to the atmosphere in the form of perspiration and respiration. This is estimated at 9.0 lb/day for the APOLLO vehicle. This H_2O, if removed from air which is filtered of bacteria, is acceptable for use as drinking or wash water. Therefore, any system which does not save this H_2O for future consumption must have the weight of the water lost included in its system weight as a penalty.

[redacted]

Three methods of controlling humidity are considered. Two methods involve the water removed from the atmosphere before it is passed through the molecular sieves for CO_2 adsorption. One method assumes that the water vapor adsorbed on the silica gel will be desorbed to space vacuum in much the same way that the CO_2 is desorbed from the molecular sieves. The weight penalty of this method is therefore 9 lb/day. The second assumes that the water will be driven from the silica gel through the application of electrical heat, and then will be recondensed on a heat exchanger. If a weight penalty of 125 lb/kw and a heat requirement of 1400 Btu/lb of water desorbed are assumed, then the electrical weight penalty of this method of humidity control is 15.4 lb. If a weight of 5 lbs is assumed for the condenser assembly, this system will weigh approximately 21 lb.

The third method of humidity control assumes that the water vapor condensed on the heat exchanger will be collected. The thermal balance of the cabin is such that some heat must constantly be removed in the air-conditioner. Since, for the desired cabin conditions, moisture will condense on the cooling coils, no weight penalty is assigned for the cooling of the atmosphere to condense H_2O. Therefore, the only weight assigned to the water removal system is 5 lbs and 5 watts for collection and storage devices.

Figure I-2-26 shows the weight comparison of these three methods of humidity control as a function of time. It is evident that the system which collects the water condensed on the heat exchanger is the lightest.

Figure I-2-30 shows that the expected range of coil surface temperature is more than adequate to maintain the humidity within the desired range. Figures I-2-27 thru I-2-29 show three systems for the collection of condensate from the cabin heat exchanger. All of these systems provide for the separation of water droplets from the air stream by impingement on a surface after they have been blown from the cooling coil. The systems are intended to collect the water, free of entrained air, so that a separate air separation process is not required.

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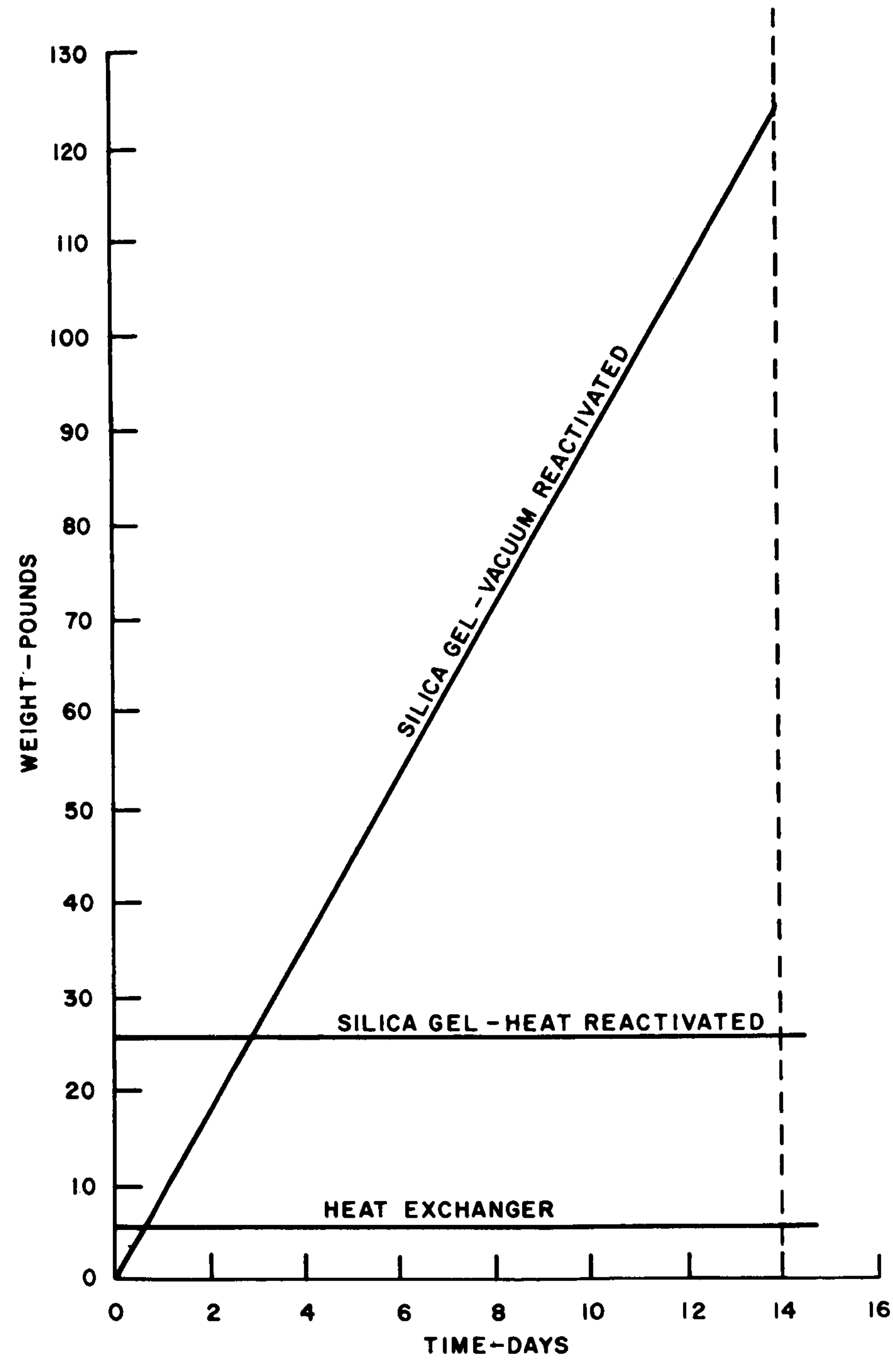
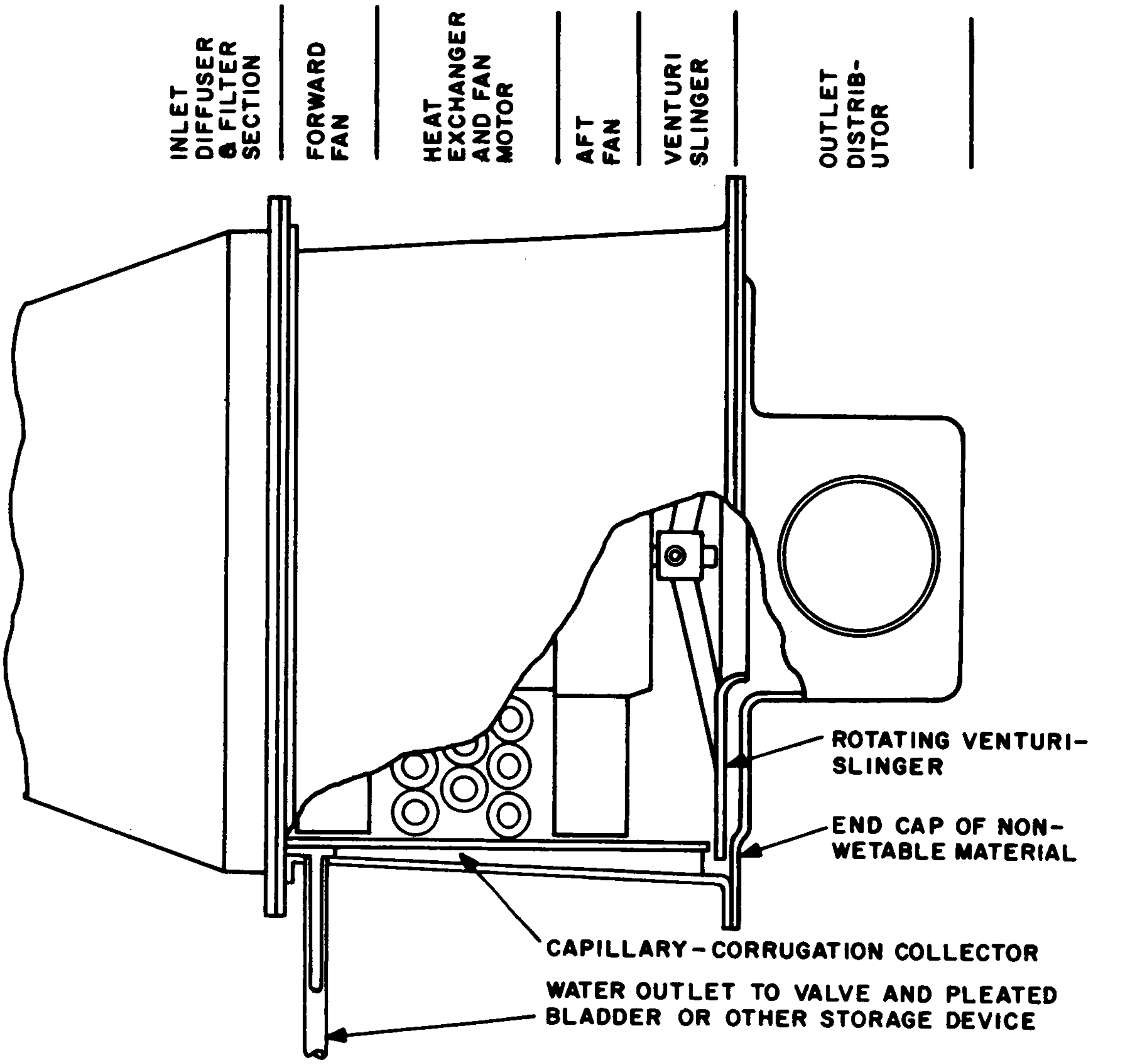
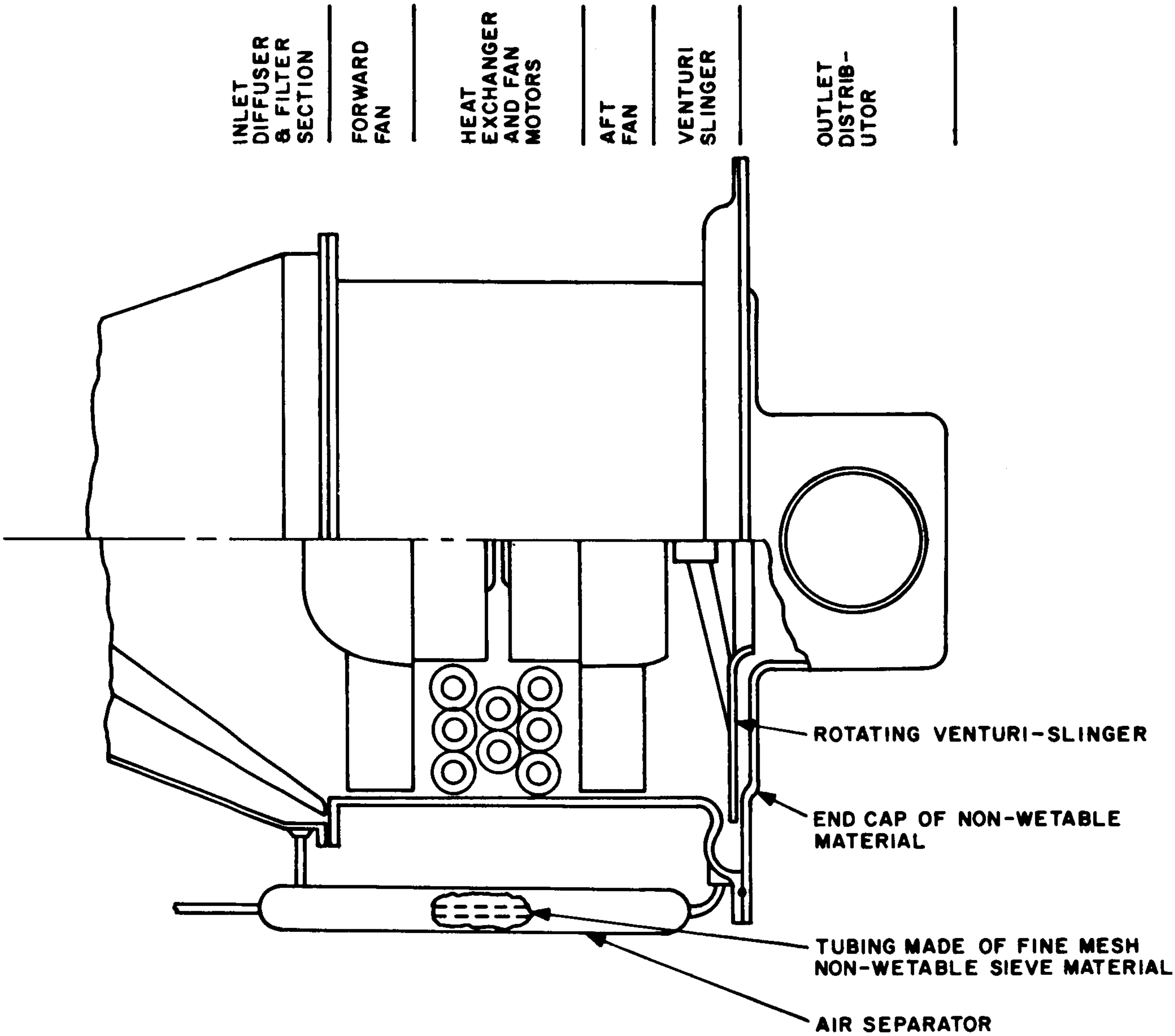


Figure I-2-26. Weight of humidity control systems versus time



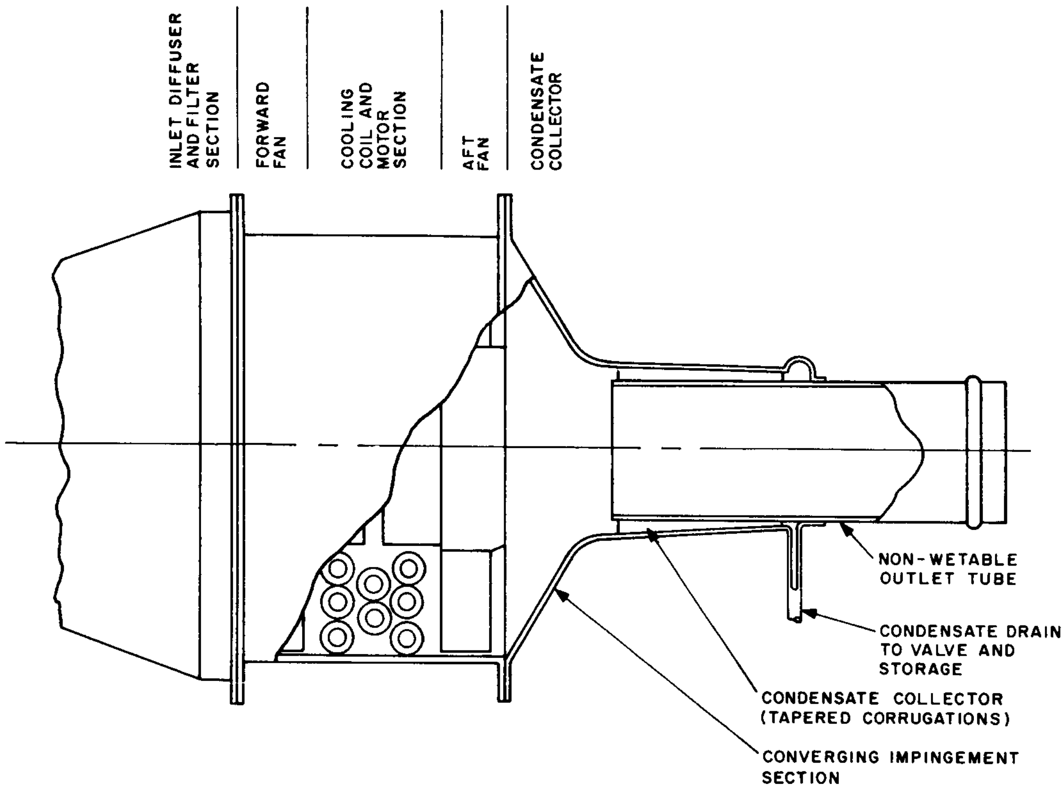
The figure shows a cabin air conditioner assembly cut away to show a zero gravity condensate collection system. Moisture from the cabin air stream condenses on the cold surfaces of the heat exchanger coil and remains there until it has accumulated in sufficient quantity to be blown off in droplets by the passing air stream. Due to their momentum, the droplets impinge on the rotating venturi-slinger, while the air stream turns and passes through the venturi. Centrifugal force throws the water from the slinger and the water wets the finely corrugated collector surfaces that are wrapped around the assembly. The corrugations are convergent. When the water films on two opposing converging surfaces meet, they form a meniscus. As additional water collects, the meniscus moves toward the divergent end of the collector.

Figure I-2-27. Slinger-Capillary water collector



The figure shows a cabin air conditioner assembly cut away to illustrate a zero gravity condensate collection system. Droplets of moisture, which have been blown from the heat exchanger coils, impinge upon the rotating venturi slinger and are thrown off by centrifugal force. The water enters a collector ring from whence it is drawn off through a small tube of fine mesh non-wettable sieve material. The sieve is surrounded by a chamber that is connected to the air stream ahead of the fan. Entrained air passes through the sieve and back into the main air stream. The water passes into a pleated bladder or other variable volume reservoir.

Figure I-2-28. Slinger-Strainer water collector



A Cabin air conditioner assembly is shown cut away to illustrate a zero gravity condensate collection system. Water, which has condensed from the air onto the cooling coils and subsequently been blown free by the air stream, impinges on the rapidly converging duct section. The water flows into the condensate collector by reason of its own wetting action and the aerodynamic forces of the air stream. The condensate collector is constructed with corrugations of metal, or other wettable material, that taper from their inlet ends down to capillary tube like dimensions at their outlet ends. When the water films that are wetting opposing sides of a corrugation meet each other, a meniscus is formed. The corrugations aft of the meniscus are filled with water (to the exclusion of air), and this may be drawn off for use in the cabin. The air outlet tube is positioned so that its inlet end is at the vena contracta of the air stream, and thus its sharp inlet neither causes additional pressure drop in the air circuit, nor creates air turbulence in the water collector.

Figure I-2-29. Impingement-Capillary condensate collector

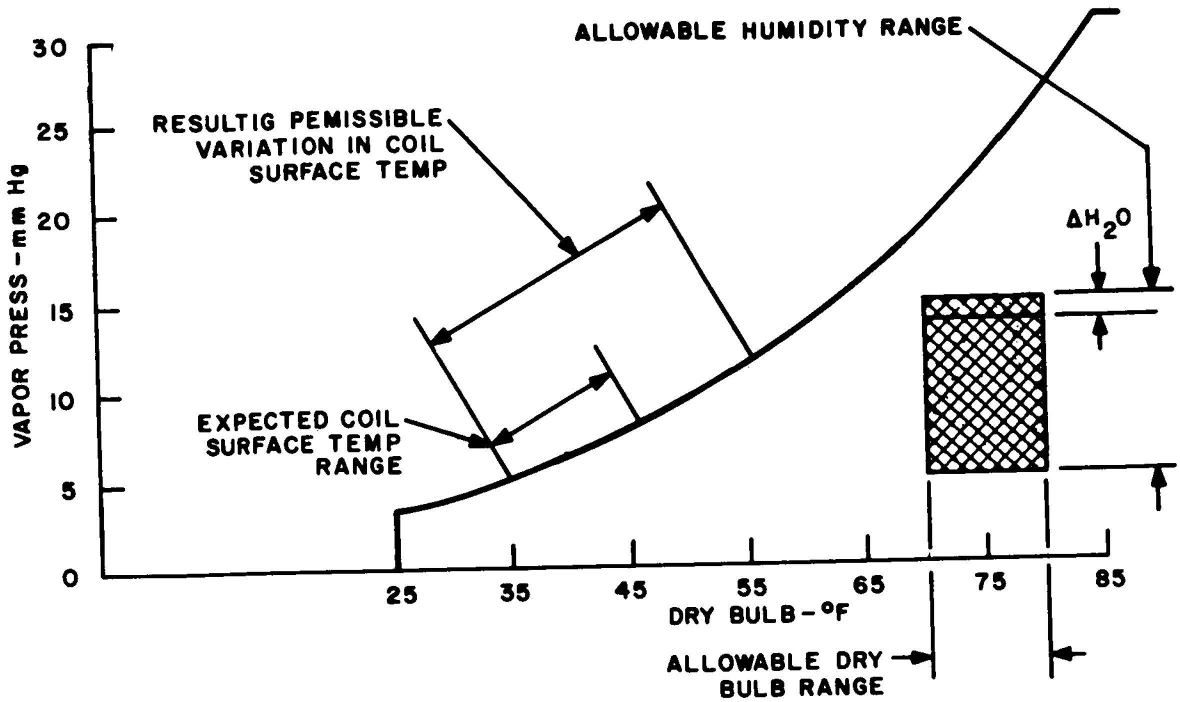


Figure I-2-30. Expected range of coil surface temperatures

2.3.3.3. NOXIOUS AND TOXIC GAS CONTROL

The potential hazard of toxic contaminants in the APOLLO cabin atmosphere must be recognized and the problems resolved. Although this problem area is expected to be of minor significance under normal operating conditions and for periods up to several weeks, it will become increasingly serious in emergency situations, as the state-of-the-art in space cabin sealing improves and as the length of the mission time increases. The unique nature of the sealed artificial environment is such that without adequate control it (a) permits the accumulation of many gases, vapors, dusts and smokes which in the natural environment appear only in trace amounts and (b) commits cabin occupants to exposure to these contaminants without interruption. The danger of relatively long exposures is obvious.

In the APOLLO application, atmosphere contaminants under normal conditions are expected to be principally in the gaseous form, although some dusting is likely to occur. Fumes and smokes may appear in emergencies such as those resulting from fires and explosions. In order to reduce atmospheric contamination, it is recommended that tobacco smoking not be permitted in the APOLLO vehicle.

The ultimate purpose of investigating the toxicity problem is to provide design criteria for artificial environments as in APOLLO which will insure occupant safety, comfort and maximum performance capability. This can be achieved through judicious selection of cabin material and equipment and appropriate contaminant filtration systems. These must come from qualitative and quantitative assays of space cabin contaminants, determination of their relative toxicities and rates of generation and evaluation of detoxification methods. An approach to the resolution of the problem is discussed in detail in Appendix HF-F.

Several methods are available for control of the concentrations of the toxic atmosphere contaminants in the spacecraft cabin. Periodic replacement of the cabin atmosphere, continuous purging of the atmosphere, catalytic combustion, and adsorption may be used singly or in combination to remove objectionable gases, smokes and fumes from the cabin. Fundamental to the control of contaminants in the cabin atmosphere, however, is the prudent selection of the materials to be used in the construction and outfitting of the cabin. The use of beryllium, magnesium, mercury, selenium, phenolics, teflon, chloroprene rubber, hydrocarbon lubricants, and volatile solvents should be avoided if possible, and any use of these materials should be under the cognizance of the persons responsible for the atmosphere purification system. Waste food, urine and feces collection devices should be designed with gas retention as a criteria.

Throughout the design program, materials should be tested for out-gassing at simulated cabin environment conditions. Additional data on the production rates and types of trace gases that are released due to the crew's metabolic processes is needed. This information can be obtained through spectroscopic analysis of the atmosphere during closed cabin habitability tests, which are expected to be conducted early in the design program to verify the selected APOLLO cabin pressure and atmosphere composition.

Although the minimum weight contaminant control system cannot be selected until the contaminant production rates are known, and can be compared against the APOLLO cabin volume and the maximum allowable concentrations, some comparison of the control methods can be made at this time. Catalytic combusion, using Hopcalite, effectively removes carbon monoxide, hydrogen, and certain hydrocarbons. Hopcalite is basically a co-precipitated mixture of equal amounts of the oxide of manganese and copper. It was developed at about the time of World War I for the purification of submarine atmospheres, for which it is presently used. Hopcalite is also used to remove carbon monoxide and hydrocarbon vapors in aircraft air cycle air conditioning systems. Activated charcoal can be used to absorb ozone, fecal odors, hydrogen sulfide, methyl mercaptan, camphor, skatol, indol, pentane, hexane, and others. Ten pounds have been included in the weight of the life support system for canisters of activated charcoal and Hopcalite. The canisters are located in the air purification circuit. Although it is not anticipated that any noxious or toxic gas, which cannot be controlled by the Hopcalite or activated charcoal, will accumulate to a dangerous level, enough reserve has been included in the cabin oxygen and diluent supply to permit the 'dumping' and replacement of the cabin atmosphere twice. Each repressurization of the cabin requires 6.3 lb of oxygen and 4.8 lb of nitrogen.

The mass spectrometer, which is to be used to sense the constituents of the cabin atmosphere during flight, will provide the signal for an instrument panel display to warn the crew of a toxic gas buildup. A cabin purge can then be accomplished to rid the cabin of the objectional gas. The mass spectrometer has the inherent capability of detecting small concentrations of gases. It can also be used to obtain broad-spectra gas analysis data which can be telemetered to the earth for analysis by a trained spectroscopist. This method is explained in more detail in a following section.

Fortunately, the problem of contaminant control is one which can be completely explored during early development phases of the program. As soon as the materials which will be contained in the vehicle are fairly well defined, a system test can be conducted. Laboratory-type instruments, such as mass spectrometers and gas chromotographs, can be used to determine precisely if the concentration of any gas — not accounted for, based on the results of previous study — builds up to an annoying or toxic level. If so, and if possible, the source of the offending gas will be removed. If removal is not possible, then a specific control mechanism can be added to the cabin environmental control system.

2.3.3.4 PARTICULATE MATTER CONTROL

2.3.3.4.1 Physiological Considerations

The presence of weightlessness, a primary problem area to all involved with space technology, forces us to regard with special caution airborne bacteria, dust, spores, enteric micro-organisms, and water droplets. Not only does the hazard of these "floating" substances result in a requirement for filtration of particulate material, but there exists the possibility that radiation sources encountered during flight will accelerate mutation rates and alter the metabolic requirements of pathogens. The astronauts will be living in a virtual incubator in which many new conditions will be created. Floating dust particles will provide vehicles for any and all pathogens which escape sterilization procedures. Studies by Loosli suggest that inhalation of dust borne bacteria is more important than direct inhalation of infectious droplets or droplet nuclei in the spread of respiratory tract infections. Figure I-2-31 shows various interesting characteristics of airborne particulate matter as a function of particle size. A point worthy of note is that the settling rate of 0.1 to 0.5 micron particles — the size range which is most harmful — is so low that almost any air motion at all will keep the particulate matter airborne. Consequently, the loss of normal gravitational precipitation, due to the zero gravity environment of the APOLLO vehicle, should not significantly increase the hazard from airborne particulate matter.

2.3.3.4.2 Precautionary Measures

Adherence to the requirements of the microbiologist during the actual design and construction of the vehicle and the cabin will alleviate failures due to deterioration, fungal growths and increased bacterial and viral counts which might hasten the breakpoint. Theoretically, man will be the major source of particulate contamination; however, the carcinogenic agents and metallic dusts from instrumentation and motors also require consideration during construction. Due to the dangers involved from overloading the filtration and system responsible for removing particulate matter, it will be necessary to control washing, shaving, and to prohibit smoking.

Numerous precautions must be taken prior to the entrance of the astronauts to the cabin. Each member will have to be free from any infection or incubating of potential airborne diseases. The crew will be required to wear specially constructed and treated clothing which will inhibit microbial growth. To further aid control of particulate matter, entry into the vehicle through a series of decontamination locks is desirable. Furthermore, it would be desirable to prevent the inclusion of infectious organisms within materials, since these organisms may be released during the flight.

2.3.3.4.2 Waste Management

Under ordinary conditions, the coliform and bacilliform bacteria in feces pose no problem. However, the cabin environment with the possible mutagenic factors of increased ionization due to cosmic bombardment, increased pO_2, pCO_2 and decreased pN_2 forces us to be more cautious with airborne pathogens.

Micropore filters interposed in the waste receptacle will retain at least 95 percent of enteric micro-organisms.

Suggested limits of bacterial contamination:

Colebrook. Cawston (1948) - 50 bacteria carrying particles cu ft

Lamanna and Mallette (1959) - 10 particles/cu ft

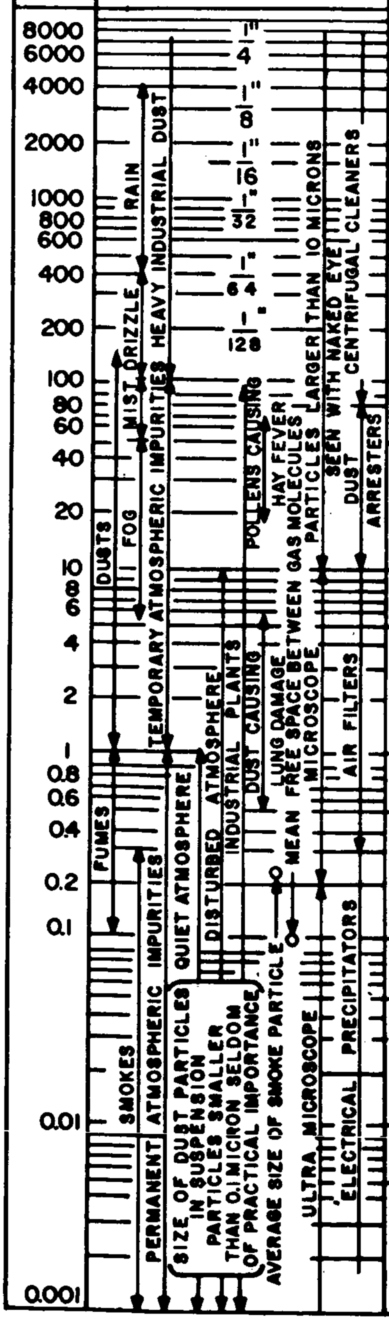
Since the early considerations of safe limits for particulate matter in the inspired air were designated, the tendency has been to accept smaller values as safe limits. It is tenable that our limits should be further reduced to insure maximum safety. The data in the above table indicate that attempts to provide sterile air are perhaps not required. A gnotobiotic environment might produce undesirable effects. Determination of the degree of sterility should be ascertained. Studies by Trexler at the University of Notre Dame on germ-free animals can serve as a convenient reference point. His studies showed that harmless airborne inhabitants of mammals may become extremely dangerous if administered to germ-free animals.

2.3.3.4.4 Particulate Matter Sources

The mechanisms of particle production are primarily:

(1) Evaporation and recondensation
(2) Chemical dissociation
(3) Mechanical generation
(4) Mechanical dispersion and evaporation
(5) Conversion of gases.

DIAM. OF PARTICLES IN MICRONS	SCALE OF ATMOSPHERIC IMPURITIES



RATE OF SETTLING IN FPM FOR SPHERES OF DENSITY AT 70 F	NUMBER OF PARTICLES IN ONE CU FT	SURFACE AREA IN SQUARE INCHES
	AIR CONTAINING 0.0006 GRAINS OF IMPURITIES PER CU. FT (DENSITY=1)	
1750		
790	0.075	0.000365
555	0.6	0.00073
59.2	75	0.00365≅
14.8	600	1/16 IN SQ 0.0073
0.592	75000	0.0365≅
0.148	600,000	3/16 IN SQ 0.073
0.007=5" PER HR	75×10^6	0.365 ≅
0.002=1.4" PER HR	60×10^7	3/8 IN SQ 0.73
$0.00007 = \frac{3"}{64}$	75×10^9	3.65≅
0	60×10^{10}	1.9 IN SQ 7.3
0	75×10^{12}	36.5≅
0	60×10^{13}	1/4 SQ FT 73.0
0	75×10^{15}	365≅ 2.53 SQ FT

LAWS OF SETTLING IN RELATION TO PARTICLE SIZE

(LINES OF DEMARCATION APPROX.)

PARTICLES FALL WITH INCREASING VELOCITY

$$c = \sqrt{\frac{2gds_1}{3Ks_2}}$$

$$C = 24.9\sqrt{Ds_1}$$

PARTICLES SETTLE WITH CONSTANT VELOCITY

STOKES LAW

$$c = \frac{2r^2 g}{9} \frac{s_1 - s_2}{\eta}$$

FOR AIR AT 70° F

$$c = 300{,}460\, s_1 D^2$$

$$C = 0.00592\, s_1 D^2$$

CUNNINGHAMS FACTOR

$$c = c^1\left(1 + K\frac{\lambda}{r}\right)$$

c^1 = c OF STOKES LAW

K=0.8 TO 0.86

c = VELOCITY CM/SEC

C = VELOCITY FT/MIN

d = DIAM OF PARTICLE IN CM

D = DIAM OF PARTICLE IN MICRONS

r = RADIUS OF PARTICLE IN CM

g = 981 CM/SEC^2 ACCELERATION

s_1 = DENSITY OF PARTICLE

s_2 = DENSITY OF AIR (VERY SMALL RELATIVE TO s_1)

η = VISCOSITY OF AIR IN POISES $= 1814 \times 10^{-7}$ FOR AIR AT 70°F

$\lambda = 10^{-5}$ CM (MEAN FREE PATH OF GAS MOLECULE)

PARTICLES MOVE LIKE GAS MOLECULES

BROWNIAN MOVEMENT

$$A = \sqrt{\frac{RT}{N}\frac{t}{3\pi\eta r}}$$

A = DISTANCE OF MOTION IN

R = GAS CONSTANT $= 8.316 \times 10^{-7}$

T = ABSOLUTE TEMPERATURE

N = NUMBER OF GAS MOLECULES IN ONE MOL $= 6.06 \times 10^{23}$

Figure I-2-31. Size properties of airborne particulate matter

2.3.3.4.4.1 Evaporation and Recondensation. Whenever the vapor pressure of a substance exceeds the partial pressure of that substance in its ambient, molecules escape from the material into the ambient. As the vapor pressure is a function of the temperature, if a volatile material is heated, molecules can escape. If the temperature of the ambient is lower, then the molecules will recombine and become airborne particulate matter. In general, particles so produced are quite small with few larger than 0.1 microns in diameter.

An example of this mechanism is the production of many airborne particles during an electrical discharge (e.g. motor brushes, relay contacts opening), when the electrode material is evaporated and recondenses into minute particles. Many particles are also formed by this process by the smoking of cigarettes.

2.3.3.4.4.2 Chemical Dissociation. The size and concentration of particles produced by this mechanism are extremely difficult to predict or control. Take, for example, the chemical dissociation of carbonyls. Carbonyls are gaseous compounds of carbon monoxide with metals. They can be detected in metallic CO_2 tanks which were sealed for long periods of time. It is suspected that they are also formed by combustion. When released into air at normal temperatures, the carbonyls react with the water vapor present to release metal atoms which form small metallic particles.

2.3.3.4.4.3 Mechanical Generation. This mechanism includes any action, either natural or man-made, by which particles are mechanically produced and distributed into the air, such as the rubbing and consequent abrasion of two surfaces (such as cloth on cloth, causing lint). The particles generated by this mechanism are physically torn from the parent material and dispersed into the air. Liquid sprays are also considered in this category. The particles so produced are generally sized from 0.1 to 100 microns.

In determining the ease of production, the mechanical properties of the material, such as crystal structure, density, and bonding forces, are important for the solids; viscosity and density are the more important factors for the liquids.

2.3.3.4.4.4 Mechanical Dispersion and Evaporation. This mechanism by which particles can be produced involves the removal or evaporation of material which bonds the particulate matter to a larger mass.

2.3.3.4.4.5 Conversion of Gases. The formation of particles from the chemical change of a gas brought about by oxidation has been demonstrated in the laboratory and in the free atmosphere. The importance of this mechanism has only recently been realized. By chemical change, gases in the air may form other compounds which tend to be solid or liquid under normal pressures and temperature. This process occurs for example, in the smogs of Los Angeles. An excellent example of this type of production is the conversion of SO_2 gas into SO_3 by reaction with ozone and the subsequent formation of H_2SO_4 droplets when water vapor reacts with SO_3. By this process, very small droplets in the range from 10^{-7} to 5×10^{-7} cm in diameter are produced.

The particle sizes which are most detrimental to men are those in the range of 0.1 to 0.5 micron in diameter. Consequently, particles in this size range must be removed with a relatively high efficiency. The filtering system will be designed so that there will not be excessive lung irritation, coughing and eye irritation due to airborne particulate matter, or an excessive accumulation of airborne bacterial spores and micro-ogranisms.

Some possible methods of filtering the air are:

(1) Centrifugation

(2) Particulate filtration

(3) Electrostatic filtration

(1) Centrifugation is used primarily to collect the coarser particles in the air. The fact that the precipitation rate, in a one g field, is so low for particles

in the 0.1 to 0.5 micron range, is evidence that centrifugation is not particularly suitable. In fact, centrifugation is not ordinarily used for particles less than 80 microns in diameter.

(2) Particulate filtration is widely used and highly effective. Table I-2-III lists some significant characteristics of some commercially available filter materials. It can be seen that extremely high filtering efficiencies can be obtained for the small particle sizes.

(3) The electrostatic filtration process charges the dust particles so that they can be collected on high-potential plates. This will probably cause some ozone production, which if significant in quantity can be highly undesirable.

From the data presented, it appears evident that the use of particulate filters will do an adequate job of removing the airborne particulate matter which can be expected in the APOLLO vehicle. Not only will the filters remove with a high efficiency the particles in the 0.1 to 0.5 micron range but they will, of course, remove the larger particles with a still higher efficiency. Vomitus can cause a real problem with any filtering system. About the only way of handling this problem is to carry a spare filter so that the contaminated filter can be replaced.

Of the filter materials listed in Table I-2-III, the Aerosolve 95 filter media has been selected for use in the APOLLO vehicle. It combines adequate filtering efficiency with relatively low pressure drop. The flat filter media will be corrugated and formed into a cone as shown in Figure I-2-32. The corrugated conical configuration was chosen to get the required 5.65 sq ft of filter area into the 12 inch diameter section ahead of the heat exchanger and fan. Characteristics of the filter are shown in Table I-2-IV.

TABLE I-2-III. CHARACTERISTICS OF FIBROUS FILTER MATERIALS

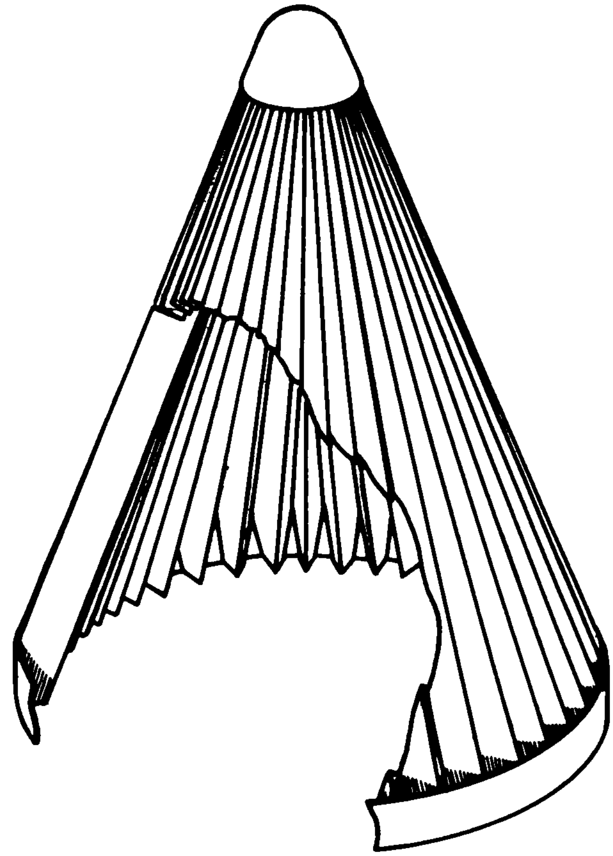
Code	Trade Name	Material	Diameter of Glass fiber μ	Efficiency %*	Rating Test	Rated Velocity, Ft per min.	Pressure Drop In H_2O !	Efficiency vs Bacterial Droplet Nuclei %	Efficiency vs Dustborne Bacteria %
H 93	CM 114	Glass-fiber paper	1.0	99.97	DOP ±	5.5	0.90	99.97	100
H 98 B	Micretain	Glass-asbestos paper	1.0	95	DOP	5.0	0.40	96.7-98.1	100
Ceramic	Fiberfrax	Aluminum silicate paper	2.4	83	DOP	5.0	0.18	97-98	99
50-FG	Filterdown	Glass-fiber mat	1.25	99-95	NBS #	21.0	0.48	87-99	99
C 95	Aerosolve 95	Glass-fiber mat	1.0-1.1	90-96	NBS	23.0	0.35	89-98	99
C 85	Aerosolve 85	Glass-fiber mat	0.8-0.9	80-85	NBS	25.0	0.22	54-69	

* During useful life of filter

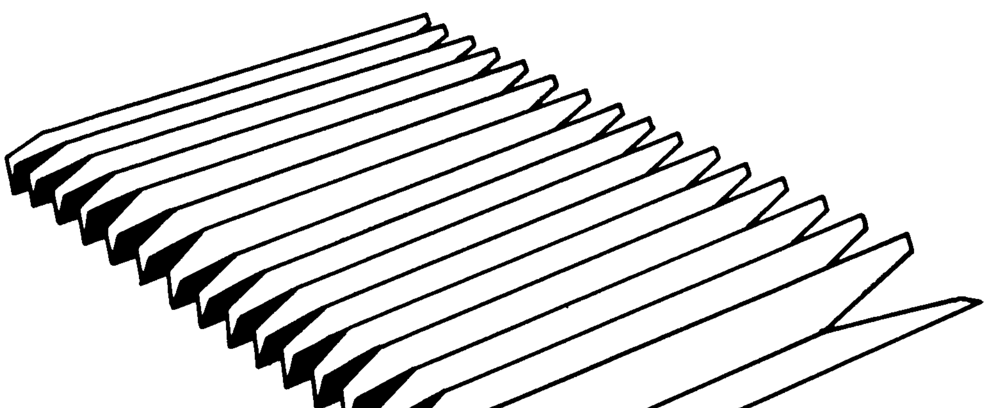
± Dioctyl phthalante 0.3 μ smoke

\# National Bureau of Standards discoloration test

! At rated velocity



PLEATED CONICAL AIR FILTER



STOWED SPARE FILTER

Figure I-2-32. Corrugated conical filter

TABLE I-2-IV. APOLLO FILTER DATA

Configuration	corrugated conical
Apex angle	60°
Outside diameter	12 in.
Flat pattern dimensions	12 in. x 60 in.
Area	5.67 sq ft
Air flow	274 cfn
Face velocity	23 fpm
Pressure drop	0.35 in. H_2O

2.3.3.5 ION CONTROL

Considerable information has accumulated in recent years concerning the effect of ionized air on biological systems. These effects range from changes in emotional behavior to alterations of enzyme systems. A good deal of the data, however, is open to question in that many of the experiments were performed under poorly controlled conditions. Yet, sufficient evidence is at hand to indicate that definite biological effects do indeed exist.

2.3.3.5.1 Sources of Ions

Any type of ionizing radiation such as x-rays, alpha particles, etc. will produce ionization. Other sources are thermionic and ultraviolet. In these two cases, electrons are liberated from metals, which in turn, may ionize the air. High voltage discharges will also produce ionization. The sizes of ions produced form a continuous spectrum, the largest being condensation nuclei surrounded by water molecules. The size and weight of an ion are measured in terms of mobility. The standard unit is the number of cm an ion moves per second in a standard field of one volt/cm.

2.3.3.5.2 Biological Effects

Present evidence suggests the possibility that the negative ion of biological importance is ionized oxygen and the positive ion is carbon dioxide. Krueger and Smith (1958, 1959), using a radioactive ion source, have studied the effects of atmospheric ions on the tracheal cilia rate of rats and rabbits. Their experiments indicate that negative ion effects occur only when a certain minimum of oxygen is available. Therc was a comparable finding for positive ions. A certain minimum of carbon dioxide was found necessary for positive ions to exert an effect.

Rinfret and Wexler (1953) report that positive ions (approximately 2,000 ions/cc.) produce alterations of the adrenal cortex. Worden (1953) reports a decrease in adrenal weight with positive ions and an increase with negative ions. Kornblueh et. al. (1958) have demonstrated the hayfever symptoms are attenuated by negative ions and that positive ions can possibly precipitate an attack of hayfever. Worden (1954) claims that negative ions attenuate hemorrhagic shock, whereas positive ions have no effect. Thus, there appears to be a correlation between negative ion effects and gluco-corticoid effects. Conversely, positive ion effects may be likened to mineralo-corticoid effects. The method and route of action of ions on the adrenal cortex remains, however, to be demonstrated.

Numerous other reports are available in the literature concerning psychological effects (negative ions cause elation, positive ions depression) and various hypotheses have been postulated concerning methods of action (vitamins, enzyme inhibition, etc.).

In view of the fact that there may be considerable ionization of the gaseous environment within a space vehicle (Van Allen and cosmic radiation, high temperature metals, etc.), it appears important to control this ionization. This may be accomplished by generating an excess of negative ions or by neutralization of positive ions. The latter method is perhaps better from a biological standpoint. Unfortunatly, however, criteria for the regulation or control of the ion density in inhabited spaces can not be delineated from the present extent of knowledge in this field.

2.3.3.5.3 Ion Generator Description

A small radiation source, such as tritium, is a promising possibility for small ion generation because of its small size, high degree of ion-generation capability, and safety. Two other methods of ion generation are corona discharge and ultra-violet radiation. These methods are not considered because of the possibility of producing ozone, which is highly toxic. A tritium source which is approved by the AEC, was developed jointly by Wesix Heater Company in San Francisco and United States Radium Corporation. It has the following specifications:

Dimensions:	Diameter	- 2 inches
	Depth	- 1 inch
	Weight	- 4 ounces (source only)
Ion generation		- 1 billion ions/sec
Radiation		- 50 millicuries; zero rems
Life of source		- 12.5 years

Power Requirements: The ion source constantly emits Beta Rays. However, in order to repel the negatively generated ions away from the source, its electrode is held at negative potential. The following power requirements are estimated to create the electrostatic field.

Voltage	approximately 400 d-c, negative
Load	10^{-8} amp or less

Power Supplies: In order to generate the electrostatic field, the following power supplies are available:

Special Van Der Grinter batteries weighing about 6 oz total with a rating of 100 volts/inch

U.S. Radium also builds their own power pack which weighs about 4 pounds and operates from 115 volts, AC

The following general assumptions are made: Cabin volume is about 300 cubic feet. Human air intake is approximately 150 cc/sec. The generator source is placed about two feet away from the head.

A tritium source in a plastic mounting is attached to an insulated base. An electrode 1-3/4 inches in diameter is placed about 1/64-inch behind the source and is maintained at negative d-c potential. The tritium source emits soft Beta particles which produce ion pairs with a radius of about 5×10^{-8} cm and a mobility of 2cm/sec/volts/cm. The small negative ions traverse the electrostatic field in the direction of the astronaut who is at zero potential. Positive ions are attracted to the negative electrode. This is shown schematically in Figure I-2-33. The average energy of the Beta particles is 5-1/2 kev. This source is capable of generating approximately one billion ions a second, with an emission distance of approximately 2 millimeters in air.

2.3.3.5.4 Other Variable Considerations

It is expected that the astronaut would be subjected to a concentration of ions in the vicinity of the head which is considerably less than at the source. Small ions disappear in transit because of positive and negative recombination, collision with larger particles and surface adsorption.

It is difficult to state with certainty what the ion concentration will be in the area of breathing influence. While it is relatively simple to generate ions, distribution problems in a cabin could be extremely complex. For example, even with an increase in density of several thousand ions per cc at the source generator, little increase of ion density may occur in other parts of the cabin because the ion concentration will be affected by other variables such as stray electrostatic fields produced by other equipments, purity of atmosphere, and air currents. In order to obtain approximate measurements of small ion concentration in the area of breathing influence, an ion-chamber mock-up would be required to simulate cabin conditions. An electrode of some potential would simulate the nose and mouth regions. Small ions would be injected into an air stream, which would approximate the capacity of human intake. Ions impinging on the electrode could then be amplified and measured by special sensitive measuring devices.

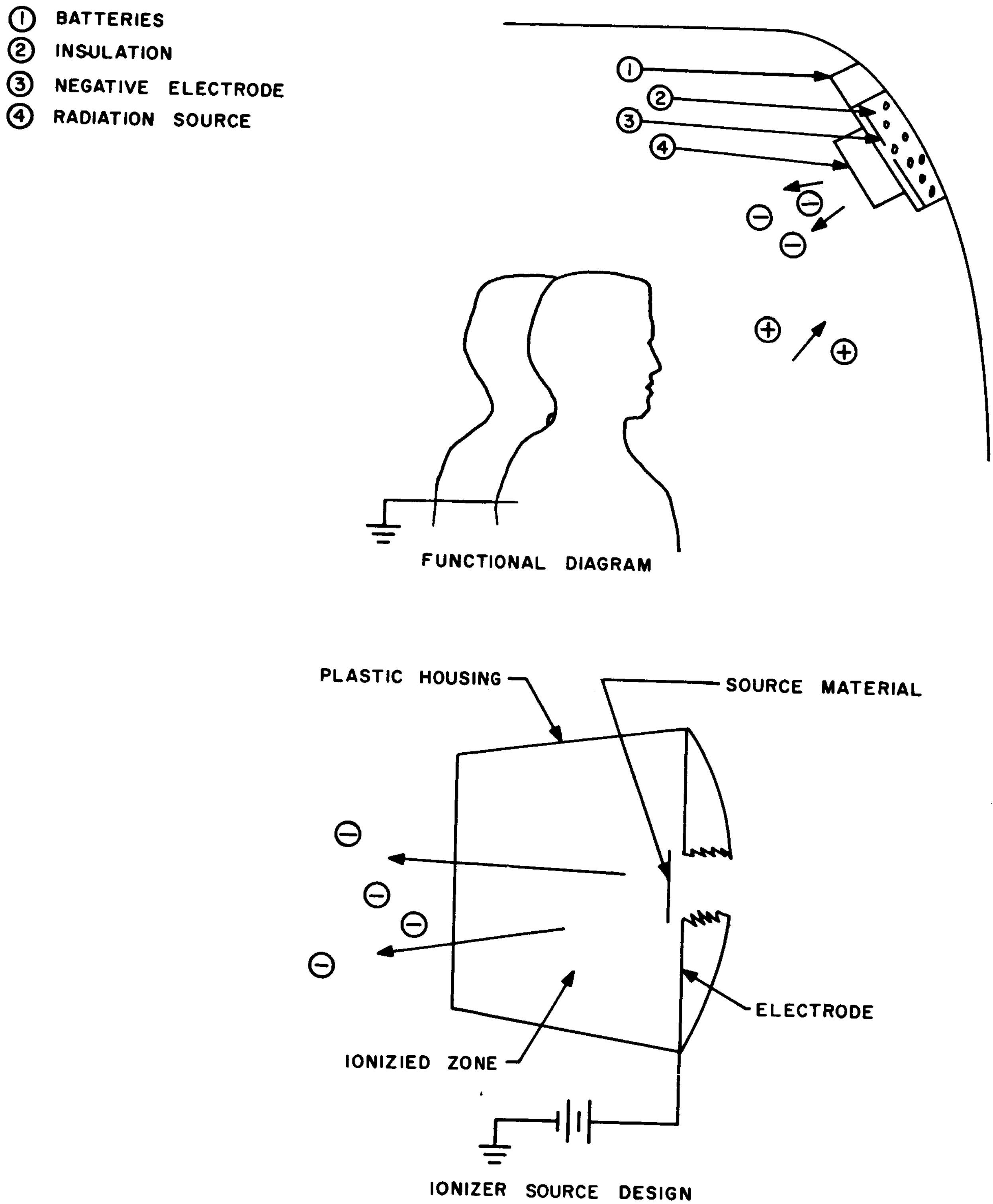


Figure I-2-33. Ionizer source design and functional diagram

Another consideration is the question of whether continuous or pulsed ion generation is necessary. Some findings indicate that the effects of negative ions may be cumulative. If this is true, continuous ion generation would be obviated.

2.3.4 Sensing and Control of the Cabin Atmosphere Composition

Several techniques have been considered for sensing the concentrations or partial pressures, of the atmosphere constituents. Mass spectrometry, infrared, magnetic susceptibility, polarographic and gas chromatography may be used to measure the concentration of one or more of the nominal constituents. Because of their nature, infrared and polarographic approaches require separate sensors for each specific gas analyzed. Gas chromatography, although suitable for multiple gas sensing, is essentially a batch process and as such requires a finite time between analyses. Although use of small-bore or capillary columns may reduce the time of measurement to several minutes, such a time delay would make a feedback system difficult to stabilize. On the other hand, a mass spectrometer approach provides multiple gas sensing plus a capability of monitoring the mass spectrum of the complete atmosphere mixture as often as several thousand times per second. This single instrument, the mass spectrometer, together with associated electronic circuitry, is the key element in the atmosphere sensing and control system of the APOLLO spacecraft.

The mass spectrometer will provide signals for cabin display of the concentrations of the most significant components, provide signals for the regulation of the oxygen and nitrogen pressures in the cabin, and can provide a signal to be telemetered to earth for command center analysis of the entire cabin atmosphere spectrum.

The APOLLO atmosphere composition is repeated here for convenience.

	Partial Pressures, mm Hg.	
	Normal	Emergency Suit/Capsule
Oxygen	180	180
Nitrogen	180	0
Carbon dioxide	0-8	0-8
Water vapor	5-15	5-15

Noxious and toxic gas can be controlled by a manually initiated purge sequence in the event that the mass spectrometer readout display indicates that the concentration of an undesirable gas is increasing to a toxic level.

2.3.4.1 SYSTEM DESCRIPTION

The proposed Automatic Control System for maintaining the desired atmospheric constituent balance is illustrated schematically in Figure I-2-34. Cabin atmosphere, bled from a point upstream of the air conditioner and carbon dioxide removal systems, is continuously available to the mass spectrometer and pressure sensor. The output of the mass spectrometer consists of a spectrum of voltage vs. mass to charge ratio. The spectrum consists of discrete voltage peaks, the amplitudes of which are proportional to the corresponding individual partial pressure of each gas present in the cabin atmosphere. The spectrum is applied to a radar-type pulse gating circuit. The gate circuit timing pulse distributes each voltage peak in the mass spectrum, corresponding to a specific gas constituent, to the appropriate analyzer network. Each specific analyzer network, one required for each gas constituent, integrates its gate input to obtain a d-c voltage whose amplitude is proportional to the concentration of the specific gas constituent. The analyzer further compares this voltage with the integrated total output of the mass spectrometer to obtain the ratio of the specific gas concentration to the total gas concentration. Since the integrated total output voltage is proportional to the total cabin pressure, the analyzer multiplies the ratio of the specific gas to total gas voltages by the voltage output signal from the pressure sensor. Each analyzer output thus results in a continuous measurement of the partial pressure for a specific gas constituent. These partial pressure output signals from the analyzers, after amplification and with proper impedance matching are used for automatic atmosphere constituent control and for cabin displays to enable the cabin occupants to monitor performance. The signals are also directed to telemetry and recording systems.

Operationally, the system functions as follows; as oxygen is consumed in the vehicle, the decrease in oxygen partial pressure (pO_2) is evident by the decrease in amplitude of the oxygen pulse from the mass spectrometer. This results in a lower signal output

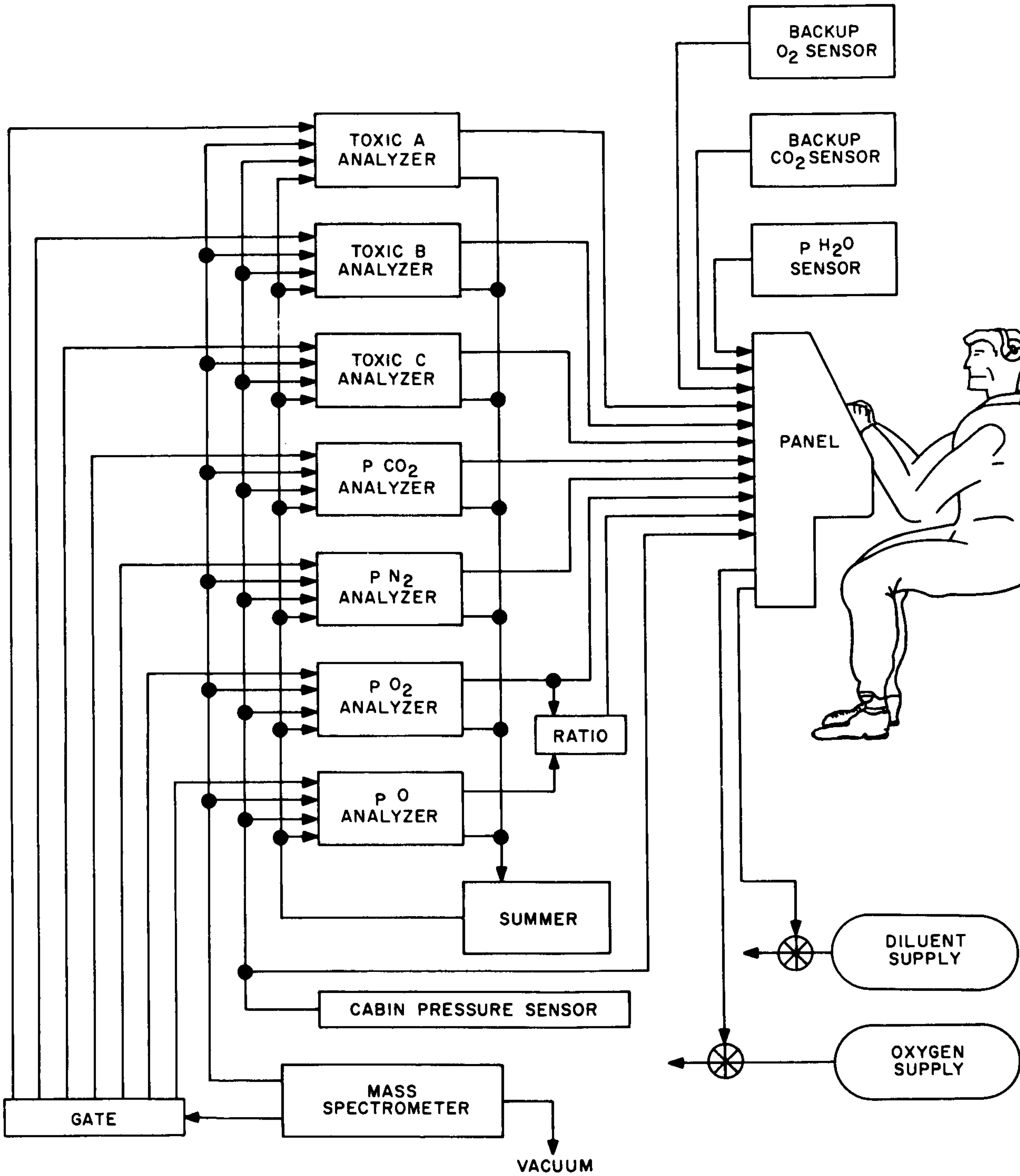


Figure I-2-34. Atmosphere sensing and control system schematic from the O_2 analyzer indicating a lower pO_2. This change in signal strength causes the O_2 control valve to open further in order to maintain the cabin pO_2 at the desired level. If the pO_2 becomes too high, a closing sequence takes place. The nitrogen partial pressure is controlled in a similar manner.

The partial pressures of oxygen, nitrogen, carbon dioxide, and selected key noxious and/or toxic gases will be displayed on the "Vehicle Condition" console in the APOLLO command module. For telemetry of the entire cabin atmosphere spectrum to a ground monitoring control center, the mass spectrometer output signal is connected to an analog strobe circuit. The output of the analog strobe circuit is a slow speed scan of the spectrum that is suitable for application to the 10 sample per second telemetry system.

No regulation of the CO_2 removal system is necessary, since the CO_2 removal system is inherently stable, its capacity increasing with increasing CO_2 concentration. The CO_2 removal systems will be designed to keep the carbon dioxide concentration in the cabin well below the permissible level.

Reference to the sections on Physiological Requirements and Noxious and Toxic gases, indicates that a great many undesirable gases can be present in the cabin at low concentration levels. The gases are produced by the crew members through metabolic processes and are liberated from the structure and installed equipment due to outgassing. Their partial pressures, which will build up during the period of occupancy of the sealed cabin, must be monitored so that detrimental concentrations can be avoided by purging them from the cabin. Since these gases are produced by (or liberated from) known sources, and the rates of production or liberation of the gases from a given source are related, it is only necessary to measure the pressure level of a relatively few 'key' gases within the cabin, in order to obtain a general picture of the complete spectrum of noxious and toxic gas pressures. Selection of these key gases must, of course, be verified for the APOLLO spacecraft during the pre-flight space environment simulator laboratory tests.

The system selected for water vapor control in the APOLLO is, like the CO_2 removal system, an inherently self-regulating process. The cabin water vapor removal system will be designed to maintain water vapor partial pressure within the specified 5 to 15 mm of Hg. without the need for regulating devices and the attendant reduction of spacecraft reliability. Water vapor partial pressure will be displayed on the instrument panel and, if desired, telemetered to earth. There are currently available a number of suitable flight type humidity sensors suitable for the APOLLO application. The electrical resistance of these sensors varies inversely with humidity.

2.3.4.2 EMERGENCY MODES

A fail-safe indicator has been incorporated into the basic system. The mass spectrometer will be calibrated initially for its mass 16 to mass 32 ratio (atomic oxygen to molecular oxygen). If subsequently this mass ratio varies beyond preset limits, a warning signal will alert the vehicle occupants to a system malfunction. Polarographic sensors for measuring the partial pressure of CO_2 and O_2 will be available for emergency use. Sealed, airtight containers will preserve their shelf and operating life. For reliability, several of these cells may be carried in similarly sealed containers and broken open as necessary. The oxygen polarographic sensor can be calibrated by making use of the diluent gas supply as a zero calibration gas and the pure oxygen gas supply as a span gas calibrant. A siphon bottle type CO_2 cartridge can be carried for calibrating the CO_2 sensor. Subtracting the sum of the partial pressures of O_2 and CO_2 derived from these polarographic cells from the total pressure given by the cabin pressure sensor will result in a measure of the partial pressure of the diluent gas present.

A new light weight oxygen partial pressure sensor is being studied at General Electric. This sensor is, in principle, a minaturized ion exchange membrane hydrogen-oxygen fuel cell. One possible version is shown in Figure I-2-35 (an excerpt from General Electric Proposal No. 031-108). The power generated by the miniature fuel cell is a function of the oxygen partial pressure in the surrounding atmosphere, and can be used to drive an indicator. A sensor of this type can be available for use as an alternate in place of the polarographic O_2 sensor since it offers greater potential reliability and utility.

During emergencies when the secondary pressurization system is in use, the atmosphere sensing and control system will maintain the oxygen partial pressure in the cocoons at 180 mm. of Hg. The partial pressures of CO_2, H_2O, and the key noxious

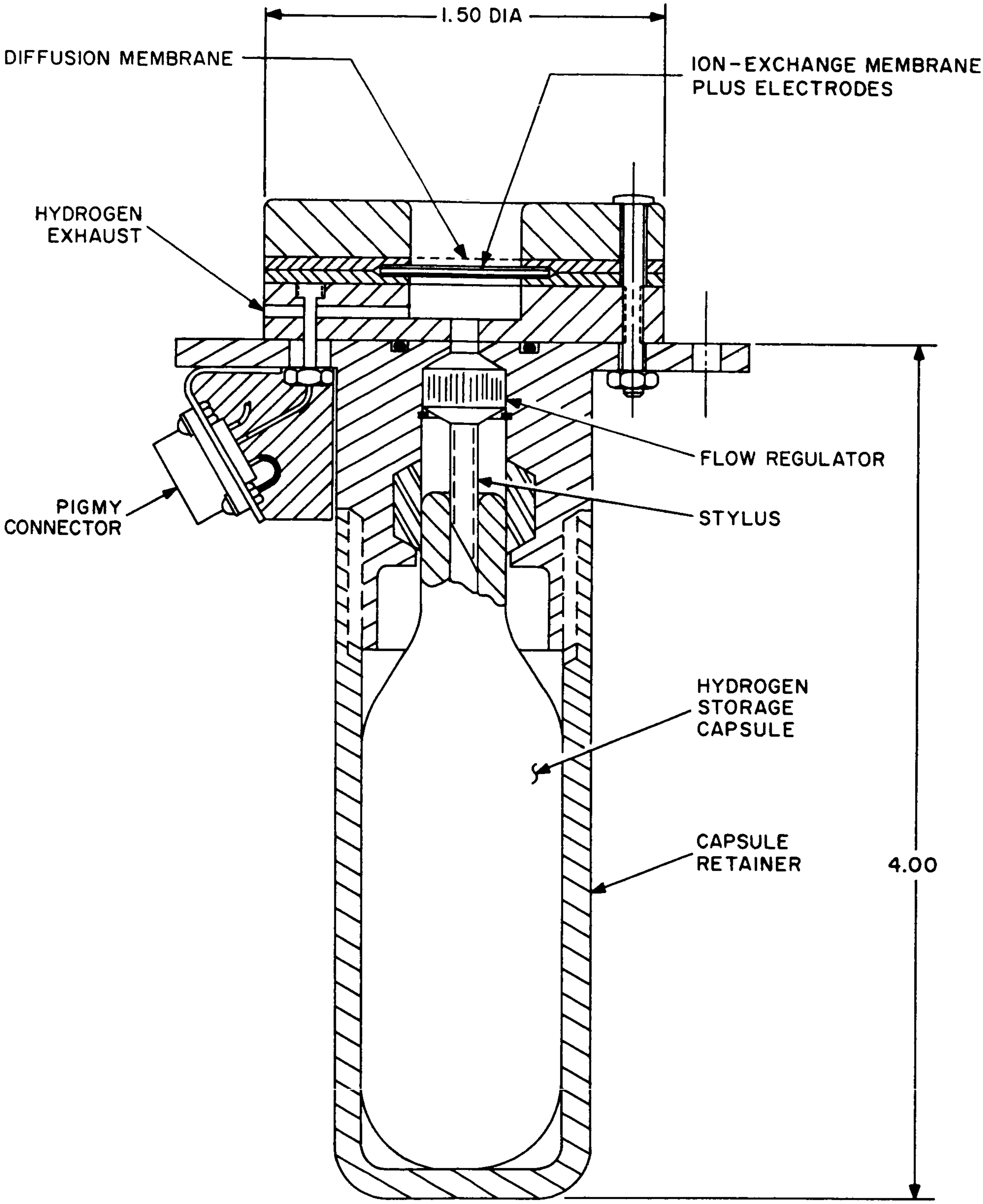


Figure I-2-35. Ion exchange oxygen partial pressure sensor packaging and toxic gases will be displayed in the normal manner. Telemetry of this information and the complete mass spectrum of the atmosphere will continue. It will be noted that the oxygen pressure selected for cocoon emergency operation coincides with that selected for the normal cabin atmosphere. This simplifies the sensing and control tasks, since it is only necessary to provide for automatic shut-off of the nitrogen supply to change from normal to cocoon emergency operation. (If the selected oxygen partial pressures had been different, simple additions to the electronic control circuitry would adapt it to the emergency mode.)

2.3.4.3 COMPONENT DESCRIPTION

As is evident from inspection of the system schematic, the key component elements are the mass spectrometer, electronic gate and analyzer modules, summer, pressure sensor and compatible flow-control valving. These key components are briefly described as follows.

2.3.4.3.1 Mass Spectrometer

Our study indicates that a time-of-flight mass spectrometer is the type of mass spectrometer applicable to our system concept, because of its versatility in being able to analyze a gas mixture for all significant gases, and because of its relative independence of drift once careful calibration has established the normalization factors for each gas for the particular instrument.

A time-of-flight mass spectrometer suitable for space vehicle applications has been developed and built by the Bendix Aviation Corporation, Research Laboratories Division in Detroit, Michigan. This instrument is an extremely compact and lightweight unit capable of delivering the complete mass spectrum of a gas mixture several thousand times per second. This feature, of course, gives essentially simultaneous and continuous data on all gases in a mixture. The required flow rate of the gas sample through this instrument to space is only 2.63×10^{-4} cc per second. The complete spectrometer weighs approximately 15 pounds and occupies a volume of less than 1/2 cubic foot. Approximately 17 watts at 28 volts is required to power the instrument.

The sensitivity of this particular mass spectrometer is better than 10 parts per million when exhausting into a conventional laboratory-type vacuum system (10^{-7} mm Hg.). For space use, the improved vacuum obtainable will result in an improvement in sensitivity, and estimated sensitivity of 100 parts per billion may be obtainable. Further gain may be realized as state-of-the-art improvements are factored into the design of this spectrometer.

2.3.4.3.2 Electronic Modules

The electronic modules include a gating module for time programming a specific voltage peak of the mass spectrum to the appropriate analyzer circuit, and a summing circuit to sum the integrated signals of the analyzers and apply this sum to each analyzer for ratioing purposes. Analyzer modules, one for each specific gas constituent, produce output signals which are used to direct flow-control valves or other equipment controlling the partial pressure of the specific gas in the cabin atmosphere. An amplifier and impedance-matching circuit are also required as part of the analyzer module. These electronic modules will be transistorized to reduce weight, volume and power requirements.

An analog strobe circuit can be used to provide a slow speed scan of the atmosphere spectrum for telemetry to earth. The function of retaining a spectrum for scanning and encoding can also be accomplished through the use of a two-speed thermoplastic recorder for fast recording and slow playback. A possibility exists for sharing the latter type of equipment with an board television system.

2.3.4.3.3 Pressure Sensor/Flow Control Valves

Either a Giannini Type 45154 or Fairchild-type pressure sensor is applicable. Both types develop an electrical signal proportional to the total pressure.

A brief survey of several commercial sources did not reveal any flow-control valves suitable for a flight application. The general type of valve required is available, however, and versions suitable for space vehicle applications can readily be designed.

2.3.5 Fire Detection and Control

The APOLLO fire detection and control philosophy is to:

(1) Provide a system to detect imminent fires, before actual burning starts.

(2) Eliminate, insofar as possible, flammable materials from the cabin as well as those that produce objectionable gases when heated. The proverb, "An ounce of prevention----" is very pertinent.

(3) Reduce fire hazard by keeping the atmosphere oxygen partial pressure low and the diluent partial pressure as high as possible.

(4) Monitor the design and installation of cabin equipment. Chart the possible causes of fire and provide protection for them.

(5) Provide both a fire-fighting procedure and extinguishers so that the fire can be quickly extinguished, should one start in spite of all precautions.

2.3.5.1 DETECTION

The products of a fire are heat, radiant energy, and products of combustion. Of these, both heat and radiant energy can possibly be used to detect the imminence of fire. Abnormal operating conditions, such as high electrical current, are indications of equipment malfunction that may be expected to result in fire.

The usual temperature range of applicability of detectors sensitive to radiant energy is above 1000 F. Those that operate at lower temperatures have inordinate response times, since thermal radiation at low temperatures is small and consists largely of infra-red radiation. The usual flame temperatures are above 1000 F; therefore, detectors utilizing radiation will not satisfactorily indicate overheat conditions.

Since the indication of an incipient fire is necessary and since the detector must be operable at a set-point temperature below flame temperatures, the combustion outputs of radiant energy and combustion products cannot be utilized. Hence, heat-sensitive elements appear to be the best method of detecting overtemperature conditions. Overload current sensors can be used to advantage to disconnect electrical equipment from the power supply before it reaches a dangerous temperature.

2.3.5.1.1 Thermometric

Heat-sensitive detectors of this type utilize the phenomena of expansion of materials when heat is added. This expansion makes or breaks an electrical circuit and initiates an electrical signal to be utilized for indication and control. The material used may be liquid, gaseous, or bimetallic. Although systems of this type can be designed adequately, they have the inherent disadvantage of sensing the temperature at a point. For this reason, their use becomes prohibitive in terms of weight since many of them must be used to adequately cover the hazard areas expected.

2.3.5.1.2 Thermoelectric

Heat-sensitive detectors of this type operate on the principle of the Peltier effect. This effect is the production of a current flow when a thermal difference is applied at the junctions of a circuit of two dissimilar metals. A fire-detection system utilizing this effect suffers from the same disadvantage as the previously mentioned thermometric systems in that it will sense temperature only at a point. In addition, the output signal is in millivolts and requires amplification to operate an indicator.

2.3.5.1.3 Thermistors

Thermistors operate from the change of electrical resistance of a substance with temperature. This substance may be metallic or a semi-conductor material. The familiar resistance thermometer utilizes a metallic resistor. The resistance-temperature characteristic for metals is an increasing function. For small temperature changes, this function is linear and the rate of change of resistance is small. Ceramics have been developed which have a decreasing resistance temperature characteristic and whose rate of change with temperature is relatively large. The major disadvantage of this type sensor is contamination of the resistance element by oxidation. This necessitates enclosure of the element in a thermal well.

2.3.5.1.4 Detection System Description

Of the three types of heat sensing elements considered, thermistor-type sensors are recommended for indication and detection of actual fire and incipience of fire.

The sensing element is one whose resistance is a decreasing function of temperature. The sensing element, upon detecting changes in temperature and rate of temperature rise, will initiate a signal to the control unit. The control unit monitors the resistance of the sensing element using a Wheatstone bridge measuring circuit in which the sensing element is one arm of the bridge, and a trip-setting adjustment resistor another. The bridge null detector is the first stage of a thermistor flip-flop circuit. As the element resistance decreases to the null-producing signal, the first stage of the flip-flop cuts off, causing the second stage to conduct. The second-stage collector current operates a relay whose contacts are used to operate the warning device. Overcurrent cutouts will be used in the electrical distribution system to automatically remove malfunctioning electrical electronic equipment from the power supply.

2.3.5.2 CONTROL

Two methods of extinguishing fires are (1) spray an extinguishing agent onto the conflagration and (2) remove the oxygen atmosphere from the area of conflagration. The latter cannot be employed when the combustible material is self-oxidized.

2.3.5.2.1 Extinguishing Agents

One of the ways in which an agent acts is to cool the burning material by absorbing the heat. Another way excludes oxygen from the combustible material and thus smothers the fire. A third method is to enter into the combustion reaction chemically and hinder or halt its progress. Ideally, the agent should be capable of efficiently extinguishing any of the types of fire which are likely to occur. In addition to this, the agent must be non-toxic and should have no adverse effect upon the atmosphere regeneration system, either in its original or in its pyrolized state. The agent should also be as non-corrosive as possible in order to avoid damage to the cabin interior and components.

Table I-2-V illustrates the relative effectiveness of some possible vaporizing-liquid extinguishing agents. This effectiveness was determined by the minimum amount of agent required to extinguish a gasoline fire. Freon 13B1 and CO_2 are decidedly more effective than the others.

TABLE I-2-V. RELATIVE EFFECTIVENESS IN FIRE TESTS

Agent	Formula	Effectiveness Relative to Freon 13B1 on a Weight Basis
Freon 13B1	$CBrF_3$	100
Carbon dioxide	CO_2	85
Freon 12B2	CBr_2F_2	67
Bromochloromethane	CH_2BrCl	45
Carbon tetrachloride	CCl_4	34
Methyl bromide	CH_3Br	31

Table I-2-VI compares the toxicity of these agents, as determined by the Underwriters' Laboratories. From these data, Freon 13B1, the least toxic, appears suitable. However, this is not the complete story. The data of Tables I-2-V and I-2-VI are for unpyrolized vapor. The toxicity of these agents was also studied at the Army Chemical Center. In these tests, rats were exposed to air containing various concentrations of agents. In one series, the undecomposed vapor was used and in another the decomposed vapors, obtained after heating the compound, were used. Table I-2-VII is a summary of results, indicating that Freon 13B1 gives off toxic vapors when pyrolysis takes place. These data indicate that carbon dioxide is the only suitable extinguishing agent of those listed.

TABLE I-2-VI. TOXICITY COMPARISON

Group	Definition	Examples
1	Gases or vapors which in concentrations of the order of 1/2 to 1 percent for durations of exposure of the order of five minutes are lethal or produce serious injury.	Sulfur dioxide
2	Gases or vapors which in concentrations of the order of 1/2 hour are lethal or produce serious injury	Ammonia, methyl bromide
3	Gases or vapors which in concentrations of the order of 2 to 2-1/2 percent for durations of exposure of the order of one hour are lethal or produce serious injury.	Bromochloromethane, carbon tetrachloride, chloroform, methyl formate

TABLE I-2-VI. TOXICITY COMPARISON (Continued)

Group	Definition	Examples
4	Gases or vapors which in concentrations of the order of 2 to 2-1/2 percent for durations of exposure of the order of two hours are lethal or produce serious injury.	"Freon 12B2", dichlorethylene, methyl chloride, ethyl bromide
Between 4 & 5	Appear to classify as somewhat less toxic than Group 4.	Methylene chloride, ethyl chloride
5a	Gases or vapors much less toxic than Group 4 but more toxic than Group 6.	"Freon-11", "Freon-22", carbon dioxide
5b	Gases or vapors which available data indicate would classify as either Group 5a or 6.	Ethane, propane, butane
6	Gases or vapors which in concentrations up to at least about 20 percent by volume for durations of exposure of the order of two hours do not appear to produce injury.	"Freon-13B1", "Freon-12", "Freon-114"

TABLE I-2-VII. APPROXIMATE LETHAL CONCENTRATION FOR 15-MINUTE EXPOSURE, PPM BY VOLUME

Agent	Formula	Not Heated	Heated
Freon 13B1	$CBrF_3$	800,000	14,000
Carbon dioxide	CO_2	658,000	658,000
Bromochloromethane	CH_2BrCl	65,000	4,000
Freon 12B2	CBr_2F_2	54,000	1,850
Carbon tetrachloride	CCl_4	28,000	300
Methyl bromide	CH_3Br	5,900	9,600

An inert gas, such as nitrogen, may be used as the extinguishing agent and would be safer even than carbon dioxide insofar as the occupants are concerned. However, it would be less efficient.

Foam fire extinguishing agents act in a manner similar to the vaporizing liquid agents. They extinguish fire by smothering and by absorbing heat from the fire. The foam-type extinguishing agents enjoy the further advantage that they will remain at the point of their application. They will not spread into a thin film due to their wetting action nor will they be shaken from the point of application due to space craft accelerations or by movements of the crew. Most of the presently available foam extinguishing agents are chemicals which, when mixed together in water, produce profuse foam of the desired consistency. The use of these water-based foams on electrical fires is undesirable, however, because they will conduct electricity. This gives rise to both immediate and post-fire short circuit problems.

A nonconducting foam extinguisher is preferable for space craft use. The development of a nonconducting foam using silicone or other such base would be warranted for the APOLLO program. Carbon dioxide would appear to be a desirable propellant because of its demonstrated high effectiveness and low toxicity (Tables I-2-VI and I-2-VII). The propellant and the base should be completely miscible in the liquid state in the extinguisher and the propellant should vaporize readily at cabin pressure upon release of the mixture from the extinguisher.

2.3.5.2.2 Oxygen Removal

An effective method of combating fires on a space vehicle is to dump into space the atmosphere from the compartment in which fire exists, thus removing all oxygen from the area around the fire. This procedure requires that the atmosphere be replaced. It also requires that the astronaut, after having been protected against the vacuum in the cabin, be mobile in order to remove the cause of the fire. Because the astronaut must protect himself before dumping the cabin atmosphere, the suitability of this scheme is dependent upon how quickly he can protect himself with the secondary pressurization system.

It is interesting to note that, in a zero-g field where natural convection is absent, the fire will tend to be self-extinguishing. The oxygen molecules required to support combustion will have to diffuse through the inert (from a combustion support standpoint) gas cap surrounding the fire, composed of the diluent gas and the products of combustion. Consequently, oxygen removal will, to some extent, be provided by simply shutting off all equipment causing forced convection.

2.3.5.2.3. Selection of Control Procedures and Equipment

Thermistors will be located at strategic points throughout the APOLLO spacecraft. They will be connected through appropriate resistance bridge and relay circuitry to the fire warning display on the instrument panel. The fire warning display, which is to be located above the electrical power distribution display, will indicate warnings from as many as 56 sensing elements. Colored lights in the power distribution display will indicate whether a circuit is receiving power from source "A" or source "B" or has been cut out by the protective device.

The fire-control system, upon receipt of a fire warning, will automatically shut down all non-essential electrical equipment, including all blowers. This will automatically stop the removal of carbon dioxide from, and the addition of oxygen to, the cabin atmosphere. The procedure may also be initiated manually by striking a single switch.

A cooling and fire extinguishing foam, consisting of carbon dioxide and some physiologically inert and non-conducting viscous liquid, will then be manually directed from an extinguisher (one of which is located in each module) to the fire. The foam will leave the nozzle of the extinguisher at a relatively low velocity, so that it will adhere to the burning or smoldering material.

Fire emergencies outside of the pressure cabin will be dealt with in a manner that is appropriate to their nature. A very minimum of electronic equipment will be mounted outside of the cabin because of the on-board maintenance philosophy that has been adopted for Project APOLLO. The space craft propellant, hydrogen, cannot burn in the vacuum of space except when combined with an oxidizer. Propulsion system malfunctions will be indicated on the command module instrument panel (as described above). Emergency procedures for the propulsion system's malfunctions must be worked out in cooperation with the propulsion system designer.

2.3.6 Leak Detection

For an appreciation of the requirements for a leak detection system aboard the APOLLO spacecraft, it is necessary to examine the nature of the leaks that may occur, and the effect of such leaks on the atmosphere supply systems.

The oxygen and diluent gas storage and supply system studies have been based on a metabolic oxygen requirement of 1.6 lb per crewman per day and a maximum design leak rate of 1000 cc/min at cabin conditions. This design leak rate is equivalent to the flow that would pass through a single hole of 1/64 inch diameter (assuming C_D = .75). The oxygen lost due to leakage through this diameter is greater than is required to sustain one of the crewmen. A fault in the pressure vessel in the order of 1/64 inch diameter is quite significant, then, since it would increase the total oxygen use rate by approximately 25 percent.

Leaks in the pressure vessel of the APOLLO spacecraft may be expected from two sources; meteoroid penetration, and structural or sealant changes resulting from launch loadings. The probability of meteoroid penetration has been studied and the results of this study are reported in Volume VI of this report. It will be noted that while the probability of meteoroid penetration is not great, those meteoroids that have sufficient energy to penetrate the command module, for example, can be expected to cause punctures of about 1 inch or greater.

Detection and repair of the larger punctures, such as those caused by meteoroid penetration, is, of course, of vital importance to the success of the APOLLO mission. Dr. Von Braun has suggested the use of a flourescent gas or smoke for detecting leaks. The suggested procedure is to stop all air circulating fans, switch off the lights except for an ultraviolet source, and release a puff of a flourescent powder or smoke. The crew would find the leak by watching the progress of the smoke through the cabin. This suggested procedure, or our adaptation of it, is excellent for use in locating these larger holes.

A detection system for indicating the existence of smaller leaks has been studied at General Electric. GE Report 60SD428 describes the feasibility study of a detection system for the more minute leaks that may occur in a spacecraft cabin. The system utilizes an ionization gauge type of low pressure sensor that is adapted from vacuum chamber pressure sensing techniques. Sensors are mounted on the outside of the spacecraft cabin to detect the presence of gas molecules outside of the pressure cabin, and thereby indicate a leak. Several sensors may be located in a grid pattern on the cabin's exterior to indicate the general location of the leak. A discussion of this system for leak detection is included in Volume VII, Section 2.2.

Flow rate and quantity remaining of the oxygen and nitrogen cabin atmosphere supplies will be displayed on the APOLLO instrument panel. The oxygen and nitrogen use rate instruments will provide indications of the existence of a meteoroid penetration.

The use of an ionization gauge leak detection system or its equivalent, may be desirable in the APOLLO spacecraft. This system would alert the crew to the existence of an emergency situation in the event of a leak in the cabin pressure shell. The leak could then be repaired by the crew to conserve the onboard oxygen and nitrogen supplies and thus avoid shortening the mission. A system having 16 sensors located strategically about the spacecraft can be installed for an estimated weight of 15 pounds, and will provide a reasonable definition of the location and magnitude of the leak. The weight of the leak detection system can be justified by determining the probability of sustaining a structural or sealant leak of greater than 15 pounds per mission above the design leak rate. Methods of leak repair are discussed in Volume VI of the report.

3.0 Thermal Control

3.1 EFFECTIVE TEMPERATURE SELECTION

Human thermal comfort is a function of three things; the heat production rate of the body (MET. value), the insulating effect of the clothing worn (clo value), and the effective ambient temperature. The effective ambient temperature is not just the ambient atmospheric dry bulb temperature, but is as well a function of atmospheric velocity, density, conductivity, and humidity, and the temperature and emissivity of surrounding walls.

As the metabolic rate and the clo value are fairly well defined for the APOLLO mission, it is possible to fairly closely bracket the optimum effective temperature range. In addition, because the atmospheric density and conductivity were automatically defined when the selection of the cabin atomosphere was made, and the atmospheric velocity is fixed by the atmospheric distribution system and the selected flow rate through the conditioning system; the only important variables left which determine the effective temperature are the wall temperature, the atmospheric dry bulb temperature, and humidity.

Figure I-3-1 shows, as a function of dry bulb temperature and humidity, the thermal comfort region, the physiological compensable region, and the intolerable region for lightly clothed persons in slow moving air and surrounded by walls at temperatures of between 65F and 90F. Also shown is the region selected for the APOLLO vehicle. The lower portion of the comfort region was selected to compensate for the relatively low convective heat transfer coefficient between the astronauts and the cabin atmosphere due to the low atmospheric density and the lack of natural convection.

Reasons will be shown later why it is desirable to maintain the APOLLO internal wall temperatures within and preferably close to the lower end of the 65F to 90F range shown.

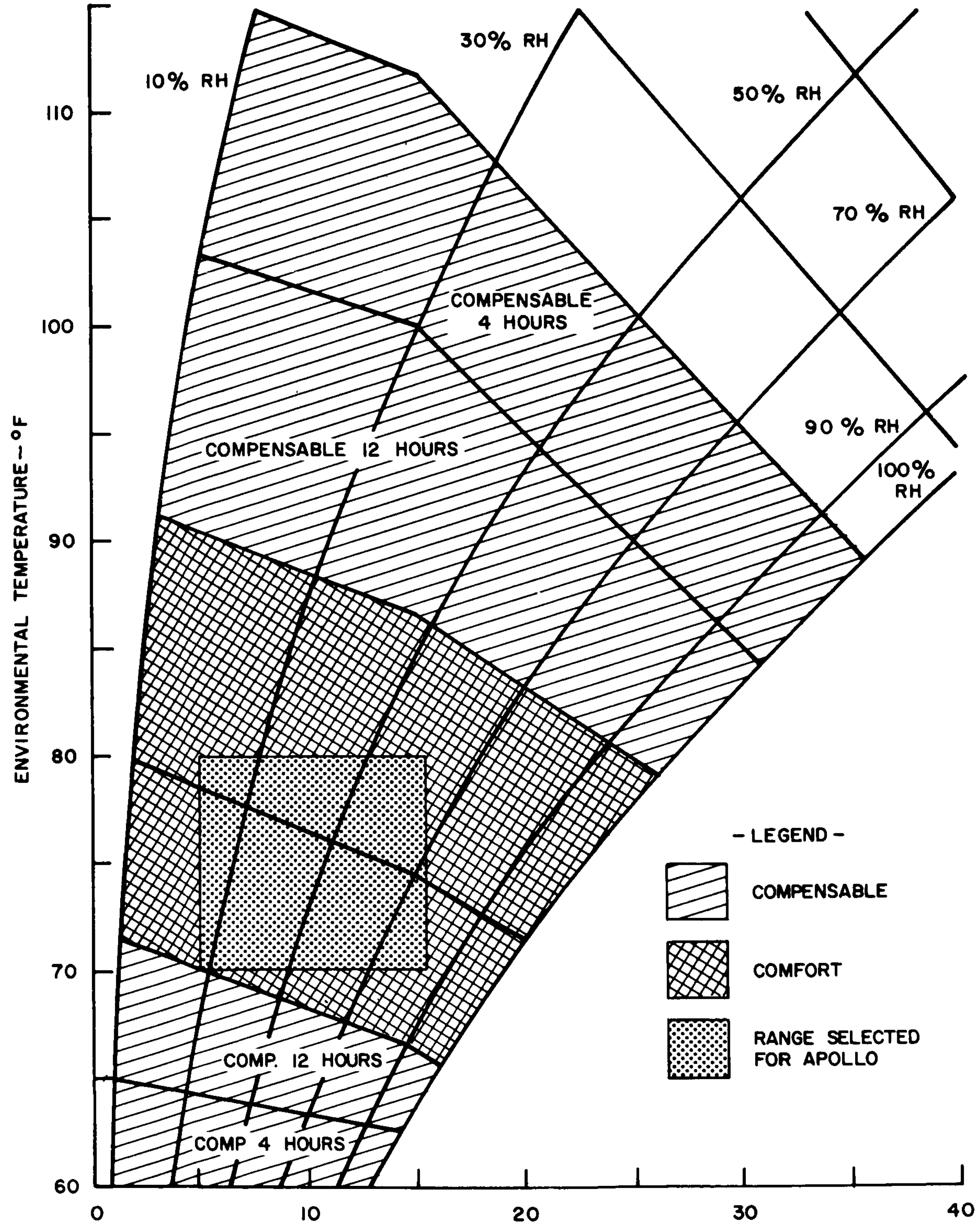


Figure I-3-1. Allowable ranges of temperature and humidity for cabins (Wall Temp, 65F to 90F)

3.2 ACTIVE THERMAL CONTROL SYSTEM

The equilibrium temperature which the internal atmosphere of a vehicle such as APOLLO will reach without active thermoregulation is a function of a great many parameters. These include such things as: the vehicle surface α/ϵ ratio; the vehicle shape; the vehicle orientation relative to the sun; the ratio of the time the vehicle is sunlit to the time it is in the shadow of the earth or Moon; the magnitude of the solar flux, which in turn depends on the time of year; the proximity of the vehicle to the earth or Moon and the magnitude of the earth or lunar albedo; the location of heat generating equipment within the vehicle; the amount of heat dissipated and the thermal resistance between the heat-generating equipment and the vehicle atmosphere and skin. Because the magnitude of each of these parameters is known only within limits, and many are variable, it is virtually impossible to design a vehicle without active thermoregulation and still maintain the atmospheric temperature within the narrow limits prescribed for APOLLO. For this reason an active atmospheric thermal control system has been selected for the APOLLO vehicle.

3.3 COMPARTMENT WALL TEMPERATURE

An Analog trace of the time temperature history of the compartment inner wall and air temperature is shown in Figure I-3-2 for the Command Module. With no internal cooling the wall temperature reaches 105 F and the air temperature 90 F at touchdown. However, due to the heat stored in the thermal protection system, it will be necessary to cool the internal compartments to prevent overheating after the command module lands (assuming still air at 70 F). This cooling will be accomplished by circulating outside air through the capsule while the vehicle is on the ground. A more complete treatment of the wall temperature of the recovery vehicle (D-2 and R-3) during re-entry is given in Heat Protection Systems, Volume VI. In the following pages, the inside wall temperature of the recovery vehicle and mission module, during cislunar flight and lunar orbit, are discussed.

Since the APOLLO vehicle is solar-oriented both during cislunar flight and lunar orbit, one-half of the vehicle is exposed to space (which has an effective sink temperature of virtually absolute zero) for almost the entire mission. Therefore, unless adequate insulation is provided, the inner wall temperature on the shaded side of the mission

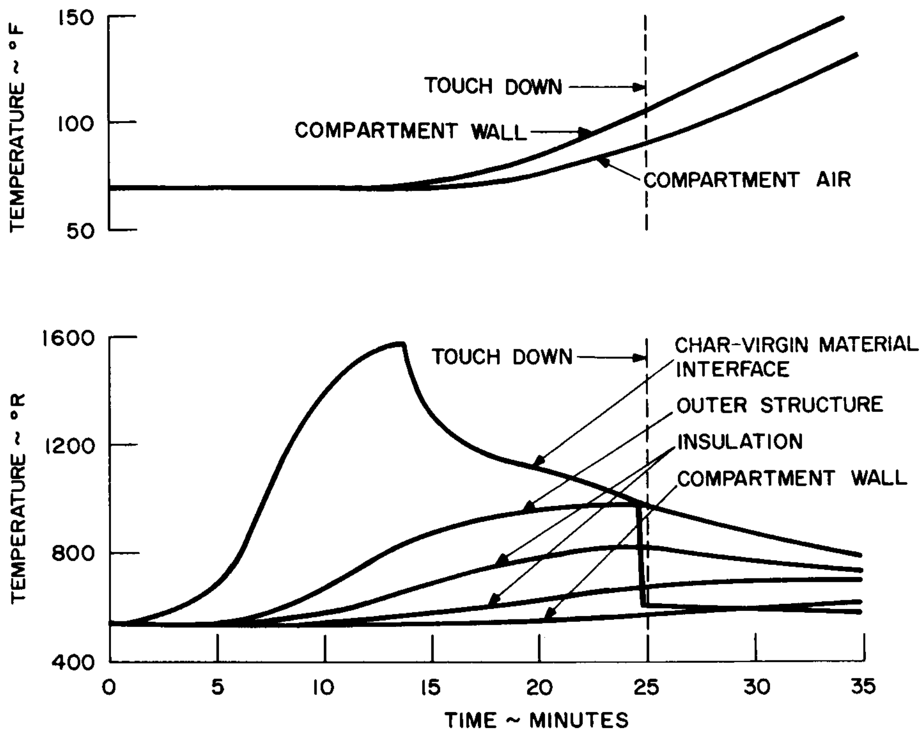


Figure I-3-2. Temperature of the D-2 recovery vehicle wall during re-entry

module will drop to a very low value. This is highly undesirable for a manned compartment since water condensation and freezing will occur on the compartment walls. To prevent condensation the compartment walls must be maintained above 64 F, the dew point temperature corresponding to the upper permissible humidity level of 15 mm Hg.

Figure I-3-3 shows the compartment wall temperature of the shaded side of the mission module as a function of the emissivity of the outer skin, the thermal resistance of the structure, and the inside wall coefficient. The inside wall coefficient is very difficult to calculate with any degree of reliability. It is a function of atmospheric velocity, atmospheric gas properties, wall temperature and length. For low velocity air flows it is difficult to separate out the effects of natural convection from the empirical data available. Natural convection will of course be absent in the APOLLO vehicle due to zero-g. However, the range shown in Figure I-3-2 of between 0.1 and 0.5 Btu/hr sq ft F should fairly well bracket the value of the inside convective coefficient that will be obtained. Consequently, it can be seen from Figure I-3-3 that, for a thermal resistance of $8 \times 10^4 \ \frac{\text{hr sq ft F}}{\text{Btu}}$, the wall temperature of the shaded half of the mission

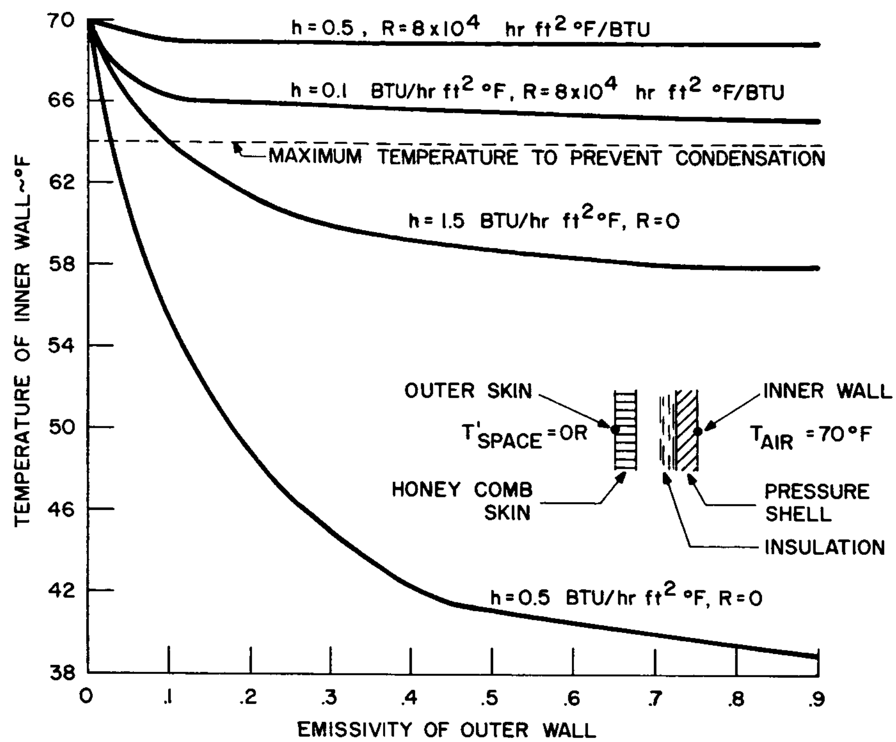


Figure I-3-3. Inner wall temperature vs outer skin emissivity for various combinations of structural resistance and inner wall film coefficients

module should fall between 65 F and 69 F. A thermal resistance of 8 x 10^4 $\frac{\text{hr sq ft F}}{\text{Btu}}$ can be obtained with approximately 0.25 inches of multiple layer thermal radiation barrier insulation placed between the inner and outer walls. This insulation is similar to the type being developed by the General Electric Company for the ADVENT communication satellite.

The wall temperature of the recovery vehicle, will, due to the high degree of insulation required during re-entry, be approximately equivalent to the internal atmospheric temperature.

The inside wall temperature on the sunlit side of the mission module can be maintained close to the inside air temperature through selective α/ϵ coating of the vehicle surface.

3.4 INTERNAL HEAT INPUT

The internal electrical heat load of the vehicle will average two kilowatts, or 6820 Btu/hr. During lunar orbit this will be increased due to the charging and discharging efficiencies of the fuel cells, as shown in Figure I-3-8. In addition, the metabolic sensible output of the crew will average approximately 855 Btu/hr and the latent output of the crew will average approximately 375 Btu/hr. This represents a total output per man of 410 Btu/hr, which is consistent with the MET level assumed in obtaining the food and oxygen requirements.

Because of the low ΔT between the atmosphere and the cabin walls, and the low internal convective coefficients, only a relatively small amount of the internally-generated heat will be transferred passively to space. Therefore, the thermal control system must be designed to carry the complete load; any heat which is passively transferred will be a safety factor to the system.

3.5 HEAT REJECTION

Two basic methods exist for rejecting heat from a space vehicle. The first is by radiation and the second is by using an expendable material, in which either an endothermic chemical reaction or physical change in state is used to provide a heat sink. The expendable systems in general are applicable for short missions. An analysis has been made by the General Electric Company of several different integrated expendable material systems, in which cryogenically stored hydrogen and oxygen are used, both as a heat sink and as fuel to generate electrical power. The most efficient of these is, for the APOLLO vehicle, approximately double the weight of a separate, non-expendable type system in which solar energy is used as the power source and radiation to space is used as a heat sink. This is due to a number of reasons:

1. The relatively long mission period of two weeks makes an expendable system heavy, because, for expendable systems, the launch weight is a function of mission time, whereas the weight of a radiation system is fixed and does not increase with time.
2. The radiator for the APOLLO mission is highly effective, due to the fact that the vehicle is solar-oriented. This causes the radiator to be both shielded from solar radiation and permits it to see space for almost the entire mission. The only portion of the mission that the radiator will receive return radiation will be for a short period during each lunar orbit. Consequently, as will be

shown later, this permits sizing of the radiator based on the assumption that the effective space sink temperature is absolute zero.

3. The honeycomb skin of the low drag fairing, surrounding the recovery vehicle and mission module, has adequate surface area to radiate all of the vehicle waste heat. It therefore has been used as a space radiator by simply incorporating light weight tubing within the honeycomb skin during fabrication. Consequently, the weight per square foot assignable to the radiator is very low.

The performance of the radiator depends upon the radiator temperature, the radiator emissivity, the magnitude of the flux incident upon the radiator and the absorptivity of the radiator to the incident flux. To more easily evaluate the effect of the incident flux upon the radiator it is convenient to use the concept of an "equivalent sink temperature". The equivalent sink temperature is the temperature of a hypothetical surface surrounding the radiator, having an emmissivity equal to that of the radiator, which would cause an incident flux upon the radiator, equal to that absorbed. It, therefore, combines the effects of solar, albedo and planet radiation into one term. Mathematically:

$$q_r = q_i + q_e$$

Since we are primarily interested in the net radiation from the radiator, or q_i, it is convenient to write the balance;

$$q_i = q_r - q_e$$

or;

$$q_i = \sigma \epsilon A (\bar{T}_r^{\,4}) - \sigma \epsilon A (\bar{T}_s^{\,4})$$

By combining terms and transposing we obtain:

$$T_s = \left[\bar{T}_R^{\,4} - \frac{q_i}{\sigma \epsilon A} \right]^{1/4} = \alpha_{r_s} (q_s + q_a) + (\alpha_{r_p} q_p) + (\alpha_{r_x} q_x)$$

where:

q_r = total heat radiation from the radiator

q_i = vehicle internal heat dissipation

q_e = absorbed portion of external incident heat flux on radiator

q_s	=	solar flux incident upon the radiator
q_p	=	planet flux incident upon the radiator
q_a	=	albedo flux incident upon the radiator
q_x	=	any other flux incident upon the radiator
α_{r_x}	=	absorbtivity of the radiator to solar flux
α_{r_p}	=	absorbtivity of the radiator to planet flux
α_{r_x}	=	absorbtivity of the radiator to any other flux
T_s	=	equivalent sink temperature
$\overline{T}$	=	radiator mean temperature
σ	=	Stephan-Boltzman constant
ϵ	=	radiator emissivity
A	=	radiator area

During the lunar orbit the equivalent sink temperature is affected by radiation from the Moon. In order to observe properly the lunar surface it is desirable that the vehicle pass over the sunlit side of the moon at low altitude. From the trajectory analysis studies, the lunar orbit which presents the most severe conditions for radiator design was chosen for investigation. This orbit has a perigee of 50 miles directly over the sub-solar point and an apogee of 1000 miles. The orbit trajectory and its characteristics are shown in Figure I-3-4.

Various points on the vehicle surface and several vehicle orientations were investigated to determine the best radiator location. For these studies the lunar albedo was assumed to be 0.07. Figure I-3-5 shows the sink temperature for different areas of the vehicle with the vehicle oriented so that its longitudinal axis is perpendicular to the orbital plane at all times.

Figure I-3-6 shows the sink temperature when the logitudinal axis of the vehicle is coincident with the Moon-vehicle radius line and the solar collector is swiveled to follow the sun. This arrangement allows the nose of the vehicle to always point toward the lunar surface for observation.

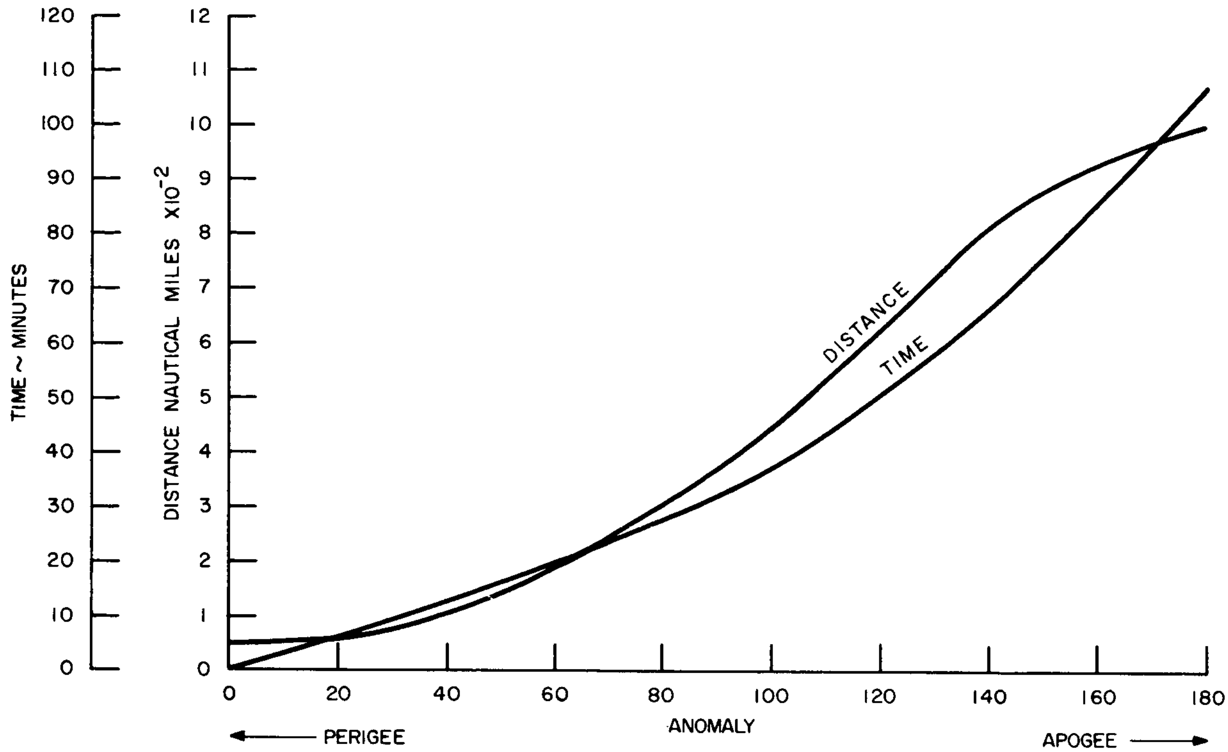


Figure I-3-4. Altitude above the lunar surface vs time from perigee for the APOLLO lunar orbit

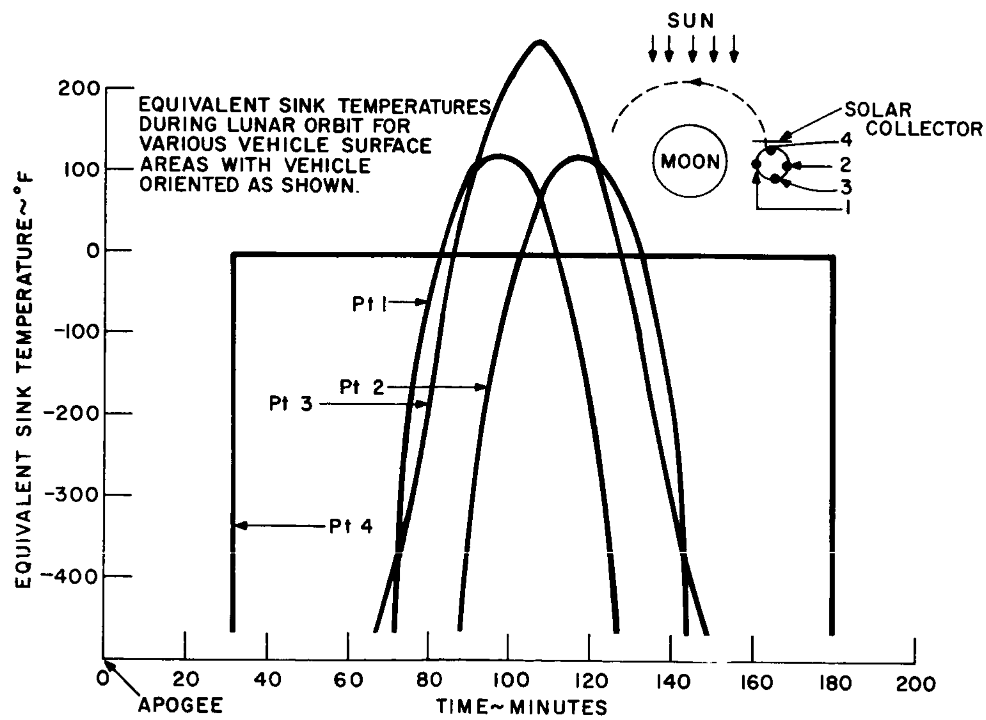


Figure I-3-5. Equivalent sink temperatures during lunar orbit for various vehicle surface areas with vehicle oriented as shown

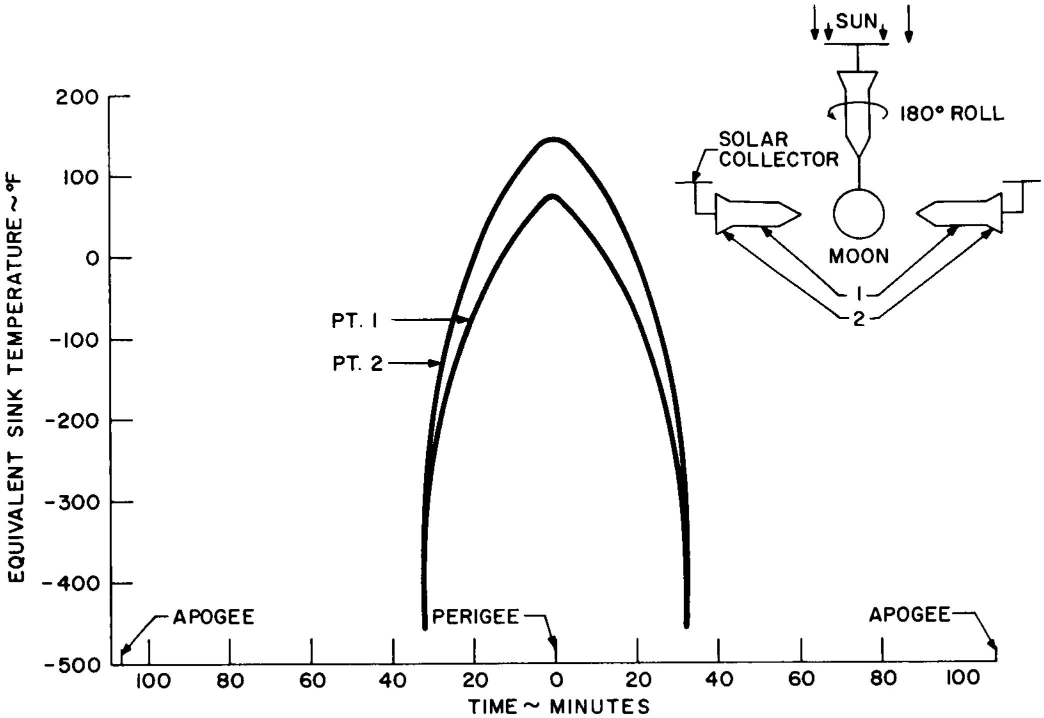


Figure I-3-6. Equivalent sink temperatures during lunar orbit for various vehicle surface areas with vehicle oriented as shown

Figure I-3-7 shows the sink temperature for a vehicle oriented as in Figure I-3-5 except that the vehicle is rotated 180 degrees, i.e., from nose up to nose down, at both perigee and apogee. This arrangement offers the following advantages:

1. The solar collector remains fixed with respect to the vehicle.
2. One side of the vehicle always looks toward the Moon for observational purposes.
3. One side of the vehicle always looks to space and has a minimum sink temperature.

This arrangement gives the minimum time during each orbit when the sink temperature is high, thus minimizing the effect of lunar radiation on the radiator performance. For this, and the reasons mentioned above, this orientation was chosen for the lunar orbit.

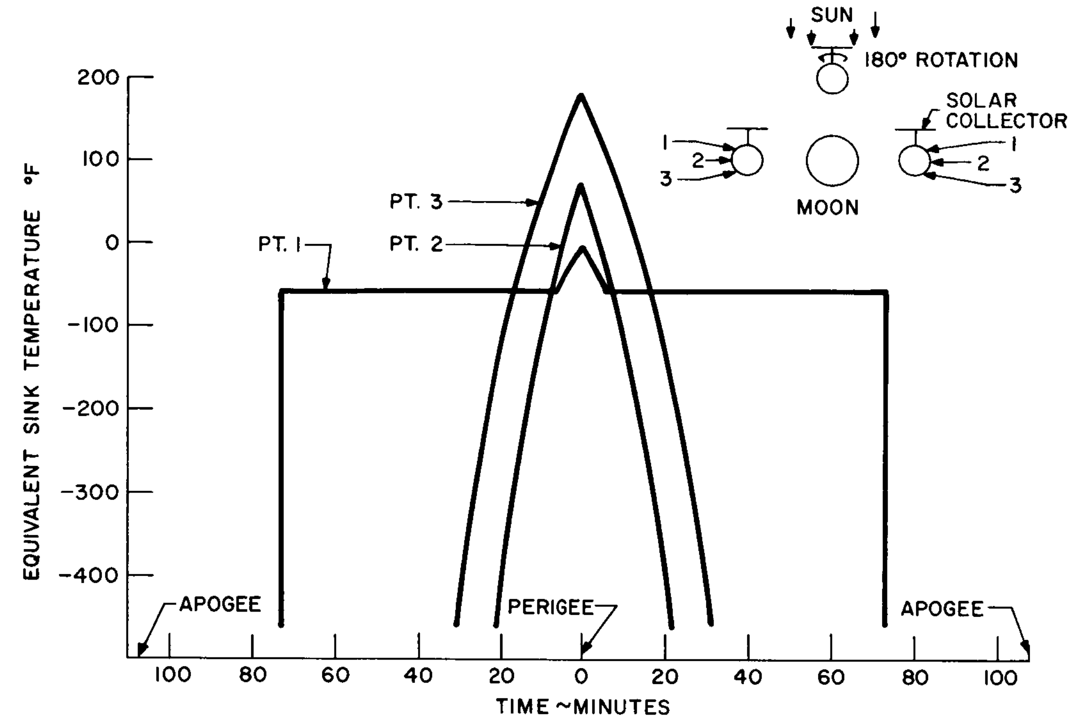


Figure I-3-7. Equivalent sink temperatures during lunar orbit for various vehicle surface areas with vehicle oriented as shown

As previously noted, the entire space vehicle will be solar-oriented during flight. Thus, during cislunar flight one-half of the vehicle is permanently shaded from the sun and has an equivalent sink temperature of approximately 0 degrees R.

3.6 RADIATOR DESIGN

As previously mentioned, the external vehicle skin is made of aluminum honeycomb, and, as the outer aluminum skin serves as an excellent fin conductor, an optimum radiator design is one which integrates the radiator into the honeycomb outer skin. In this manner only the weight of the tubing and fluid is charged to the radiator weight.

The radiator heat load is shown in Figure I-3-8 for the lunar orbit. The step function is due to the difference between the charging and discharging efficiency of the fuel cells. During the cislunar flight the radiator heat load will be somewhat lower. Part of the internally generated heat is dissipated indirectly, via the cabin air, into the radiator coolant medium. The remainder, primarily from the electronic equipment, is dissipated directly into the radiator coolant medium. (A more detailed discussion of the

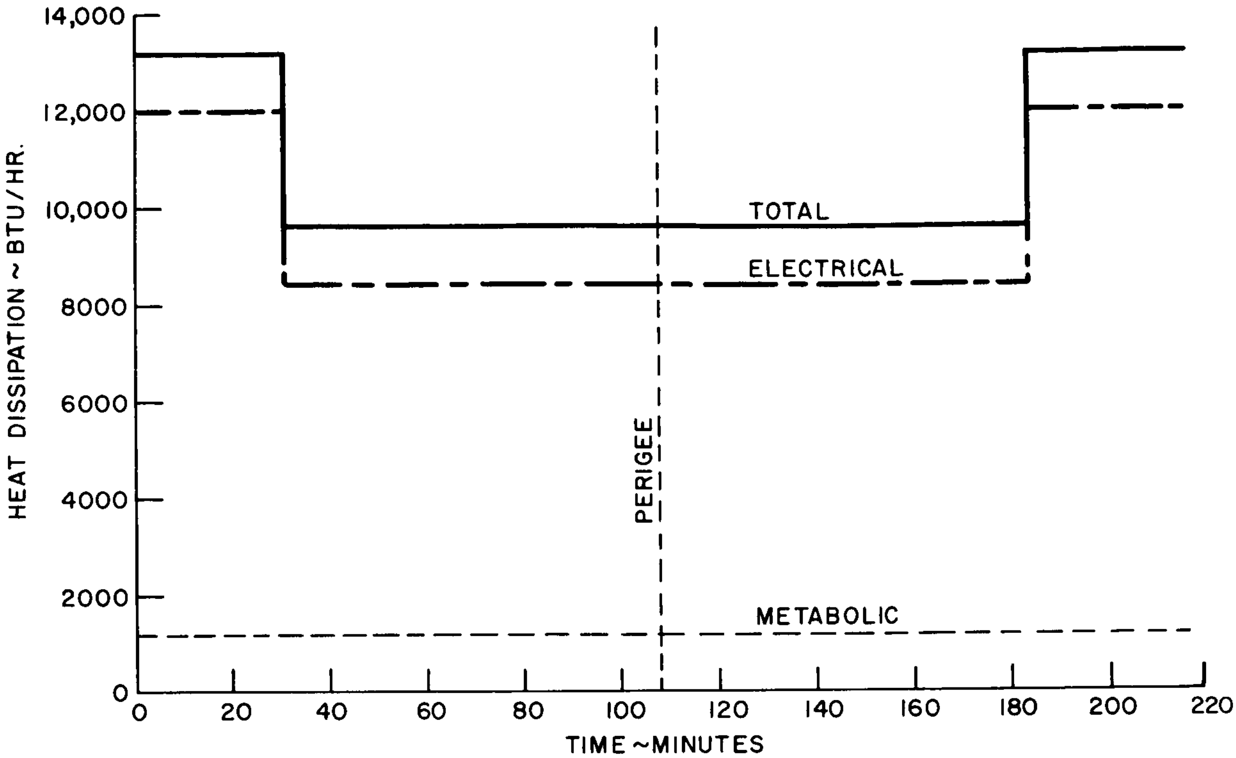


Figure I-3-8. Vehicle internal heat dissipation during lunar orbit

heat transfer cycle appears in the following section.) This allows the use of a series cooling system, i.e., it permits cooling the cabin to 70 F and the electronic equipment to 110 to 120 F with a single, continuous, heat absorbtion and rejection circuit. Thus, the radiator inlet temperature can be raised and the radiator performance enhanced. For the radiator design study a radiator mean temperature of 512 degrees R was used.

Since the design of the radiator does not have to include the weight of the radiator fins (already counted as structural skin) it appears at first that the radiator should be designed to have a low fin effectiveness, η, to reduce the total tube weight. However, during the lunar orbit the sink temperature varies for different areas of the vehicle as shown in Figure I-3-7. Since the amount of area with the lowest sink temperature is limited, a relatively high value of η results in a lighter radiator because of the increased performance of the prime area.

Using an emissivity of 0.9, an η of 0.9, and an area of 170 sq ft (area represented by points 1 and 2 on Figure I-3-7) the maximum sink temperature at which the radiator can maintain the required heat dissipation rate of 9,600 Ttu/hr is 416 R. Thus, for a period of 14 minutes of each lunar orbit, the radiator heat dissipation must be augmented. The amount of augmentation required however is small, approximately 1250 Btu. This can be absorbed by the thermal capacity of the internal equipment and structure, giving a 2 to 3 degree F temperature fluctuation. It is possible to include an expendable water cooling system, similar to the system developed by the General Electric Company for the Discoverer bio-satellites, to prevent this rise. However, it would require 8 pounds of water per day and is not considered necessary.

The radiator designed for the APOLLO vehicle has two independent circuits, as shown schematically in Figure I-3-9. If one circuit is punctured by a meteoriod, the other circuit will continue to operate, using the same radiator area. This will drop η from 0.9 to about 0.7, thereby decreasing the maximum sink temperature (to which the radiator can reject the 9,600 Btu/hr during lunar orbit) to 374 degrees R. This can be handled by either letting the vehicle temperature rise several more degrees or by decreasing the internal heat input about 20 precent, achieving the same performance as was obtained with both radiator circuits. During cislunar flight one radiator circuit will provide adequate radiation capacity.

3.7 HEAT TRANSPORT SYSTEM

A heat transportation system is required to collect the heat within the vehicle and carry it to the radiator. This can be done in one of three ways:

1. The heat can be pumped from the temperature at which it is collected within the vehicle to a higher radiator rejection temperature by means of either a Brayton or a vapor compression cycle.
2. The heat can be rejected from the radiator at a lower temperature than the collection temperature, through the use of a Rankine cycle.
3. The heat can be transported directly, via a heat transport medium, and rejected from the radiator at the same temperature as the collection temperature.

Each of these systems has its advantages and disadvantages.

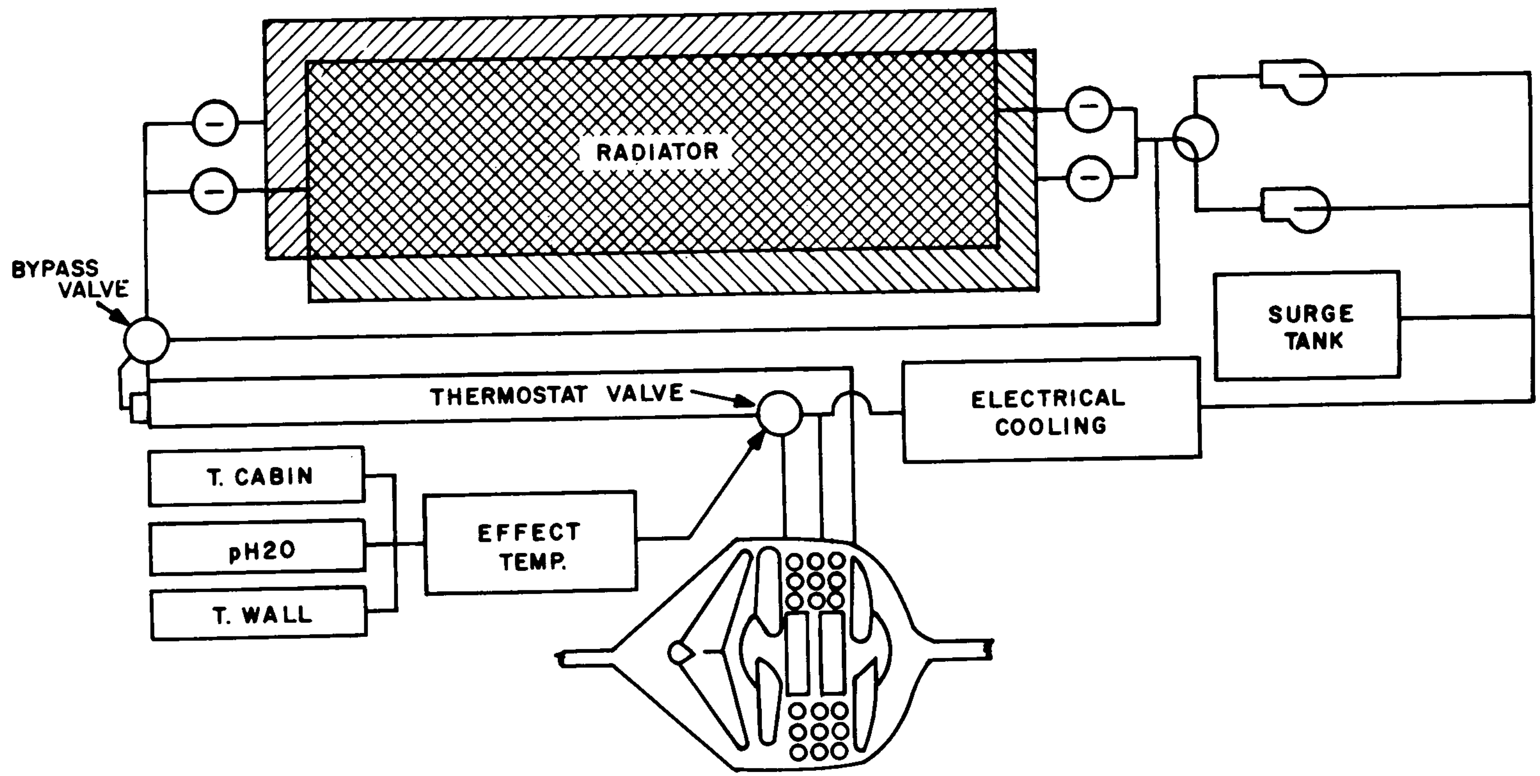


Figure I-3-9. Thermal control system schematic

The advantage of heat pumping lies in the fact that either the radiator size can be reduced oř heat can be rejected to a relatively high equivalent sink temperature. The disadvantage of heat pumping is that power is required to pump the heat to the higher rejection temperature. For the APOLLO vehicle, where the specific weight of the power supply is relatively high and both the radiator weight per square foot and the equivalent sink temperatures are low, heat pumping unnecessarily adds to both the complexity and weight of the heat transport system.

The Rankine cycle, which can be used to reject heat at a lower temperature than the collection temperature, consists of an evaporator (the internal heat collector), a gas turbine, a condenser (the external radiator), and a liquid pump. The advantage of this system lies in the fact that more work can be extracted from the gas, by expanding it from the evaporator pressure to the condenser pressure, than is required by the pump to repressurize the liquid. Thus, power can be extracted from the system. The disadvantages of this system are that a relatively large external radiator, a turbine, and a low equivalent sink temperature are required. For APOLLO this system is not advantageous because the amount of energy which can be extracted from the system does not pay for the added system complexity and weight of the turbine.

The most applicable heat transport system for APOLLO is, therefore, the simple system in which the heat collected in the vehicle is carried, via a suitable heat transport medium, to the external radiator, where it is rejected at the same temperature at which it was collected. As no physical phase changes are encountered (as is involved with the Rankine cycle), the problem of zero-g separation of gas and liquid phases, is avoided. Appendix HF-M contains a more detailed analytical comparison of the different cycles discussed.

3.8 APOLLO THERMAL CONTROL SYSTEM DESCRIPTION

The thermal control system devised for the APOLLO vehicle is shown schematically in Figure I-3-9.

The compartment atmosphere is passed through the compartment heat exchanger at a fixed rate by one of the two blowers shown. Two blowers are provided for redundancy.

The atmosphere while passing through the heat exchanger, is cooled to about 45 F and dehumidified to a dew point of about 42 F.

The heat exchanger is made from a spined fin-and-tube surface developed by the Room Air Conditioner Department, General Electric Company. This surface, due to the unusually high amount of turbulence created by the air in passing through the fins, has exceptionally high performance.

The heat is rejected within the heat exchanger to a liquid coolant which flows through the heat exchanger, then through the chassis of the electronic equipment, where it absorbs the major portion of the electrical heat input. The coolant is then pumped, by one of the two pumps shown, through the radiator, where the heat absorbed in the cabin is rejected to space. Two pumps are provided for redundancy.

A desirable feature of this system is that various components rejecting heat can be located at different places within the vehicle, without penalizing the efficiency of the system with other than the small weight of the interconnecting tubing.

The bypass valve shown in the coolant circuit is controlled by the compartment thermostat. It directs the flow of coolant either through the compartment heat exchanger or through the bypass, thereby maintaining the cabin effective temperature at the set level. There will always be a flow of coolant through a part of the compartment heat exchanger, as shown, by a parallel circuit. This insures humidity control. The cooling capacity of the parallel circuit will be just adequate to remove the minimum net internal heat input to the cabin.

The mixing valve, shown in Figure I-3-9, short circuits just enough fluid past the radiator so as to maintain the coolant temperature leaving the mixing valve and entering the heat exchanger at 30 to 35 F. A disadvantage of regulating the inlet temperature to the heat exchanger by this method is that, as the total amount of heat rejected by the radiator falls below the design valve, the heat transport fluid passing through the radiator becomes correspondingly lower. Calculations, however, indicate that even if, with a 0 degree R equivalent sink temperature, the heat load falls to 60 percent of the design load, a water -glycol heat transport fluid can still be used. If further calculations indicate that under some combination of conditions it is possible for the net internal heat input to fall below 60 percent of the design value, then another heat transport fluid, such as FC-75, can be used.

An artist's rendering of the compartment heat exchanger designed for APOLLO is shown in Figure I-3-10. As mentioned above, the two blowers are included for redundancy.

The fluted annular duct shown behind the heat exchanger in Figure I-3-10 is one of a number of methods considered by the General Electric Company to collect the condensate. Water condensed on the heat exchanger surface is blown off the coil and impinges upon the rapidly converging duct behind the heat exchanger. It then flows along the duct wall where it is collected in the corrugated flutes shown. The flutes converge so that the water films, moving down the opposing surfaces of the flutes, meet and form a meniscus. The condensate then flows into the annular collecting duct shown at the ends of the flutes. As the flutes fill with water the meniscus will move forward towards the heat exchanger. This should serve to keep the collecting ring and rear end of the flutes full of condensate and exclude air bubbles. The condensate is then intermittently pumped to a reservoir for drinking purposes. Additional methods of condensate collection, plus a description of the filter shown at the inlet to the heat exchanger, are given in the previous chapter.

3.9 GROUND AND POWERED FLIGHT COOLING

Ground cooling of the vehicle prior to launch can be readily accomplished by chilling the external radiator surface from a ground support cooling system, or by connecting a portable chiller unit directly into the fluid lines, thereby by-passing the radiator completely.

Because of the relatively low velocities in the denser portion of the atmosphere during launch, the high emissivity, and the low thermal diffusivity of the external skin, powered flight heating will not pose a serious problem. However, the radiator will not be able to reject any internally generated heat until it first radiates the heat it absorbed during powered flight. Therefore, the thermal capacity of the internal mass of the vehicle will be used to absorb the internally-generated heat until the radiator can take over. This may require pre-chilling the internal mass of the vehicle (to about 65 F) prior to launch.

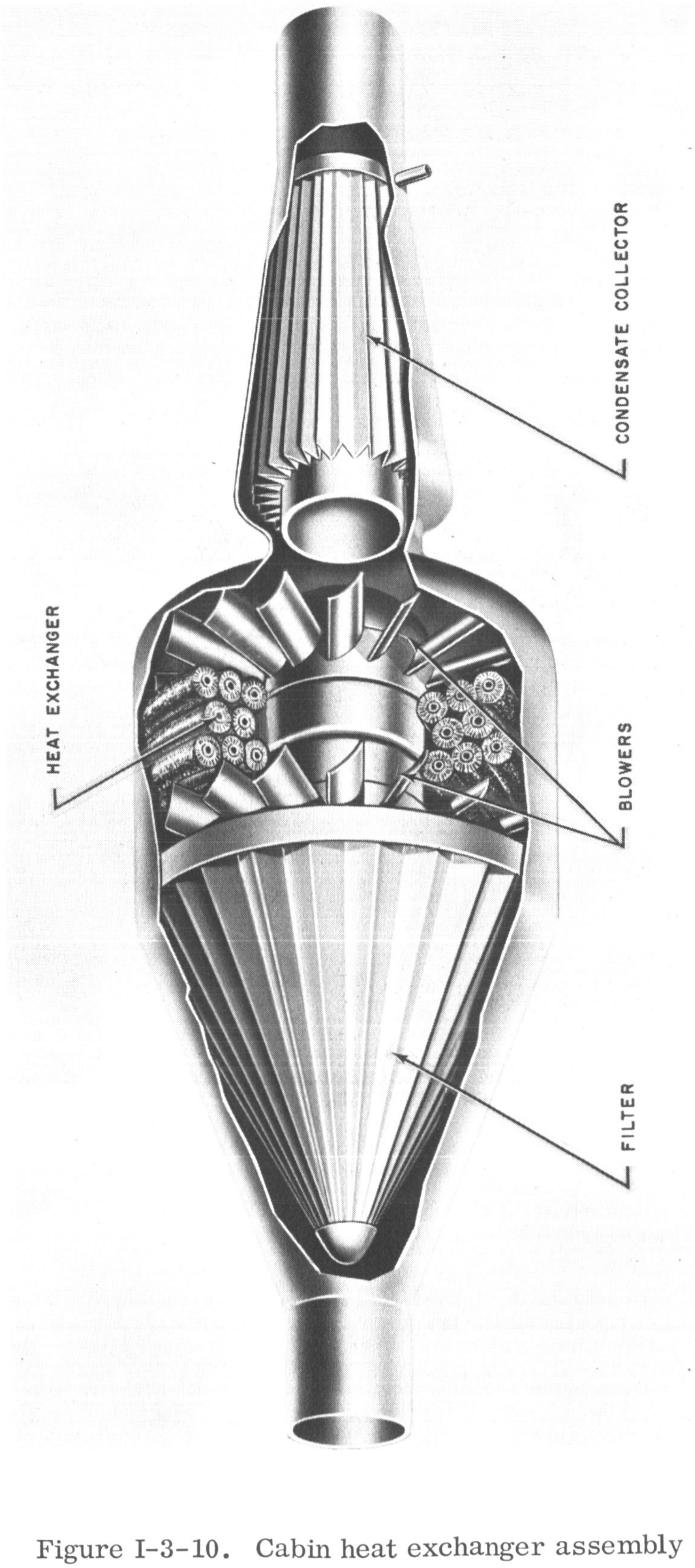
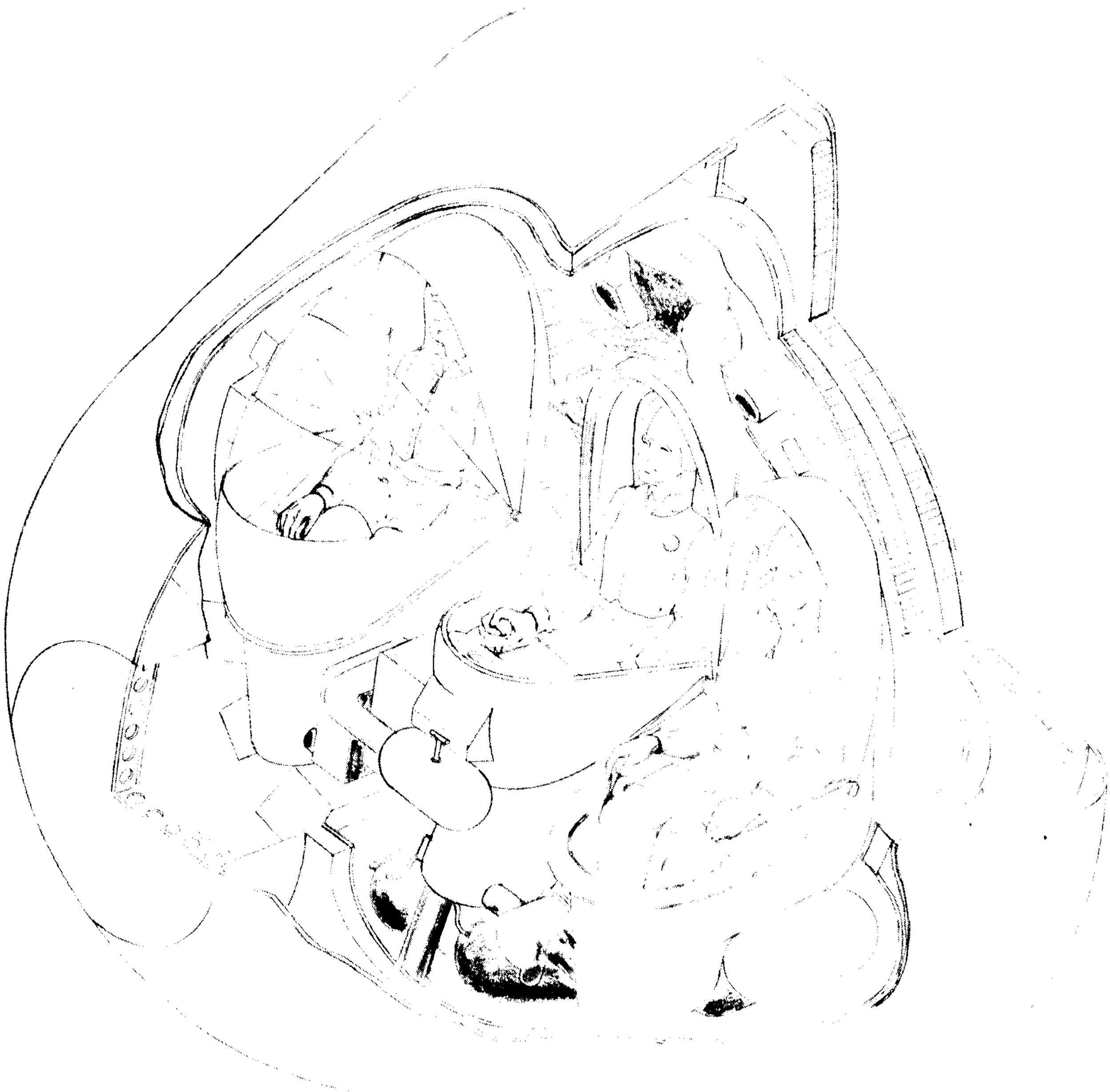


Figure I-3-10. Cabin heat exchanger assembly



Cutaway view of re-entry vehicle showing seat and cocoon arrangement

4.0 Acceleration, Vibration, Noise and Secondary Pressure Protection

4.1 ACCELERATION ENVIRONMENT

4.1.1 Acceleration Nomenclature

Examples of current nomenclature for acceleration studies are given in Table I-4-I. For the purposes of the discussions which follow, the AMAL nomenclature will be accepted and used. Note that the signs of the various symbols describing the maneuvers and physiological displacements vary in some instances.

Two vehicle shapes which have been investigated in detail during this program by the General Electric Company are shown diagrammatically in Figure I-4-1. The D-2 configuration shown is the semi-ballistic re-entry vehicle finally recommended for the APOLLO missions. Of interest here is the occurrence of positive lift at negative angles of attack (α). These negative angles of attack are typical of the vehicle's attitude during the steeper path angle (γ) re-entries and, as a result, have required positioning the man with his head toward the flap to assure a $-a_z$ (eyeballs down) rather than an a_z (eyeballs up) G-vector during this phase of flight.

During re-entry in the R-3 configuration the man is identically orientated with respect to the G-vector as in the D-2 shape. This has been done in order that the major G component is along his a_x axis rather than along the less tolerable a_z axis as would have occurred had he been placed perpendicular to the longitudinal axis of the vehicle.

Other nomenclature pertinent to Figure I-4-1 is given in Tables I-4-I and I-4-II and in Figure I-4-2.

4.1.2 Acceleration Tolerance and the APOLLO Mission

The most significant acceleration problem by far in the APOLLO mission is that of re-entry at superorbital velocities following the lunar flight. The re-entry corridors for lifting and non-lifting vehicles are established elsewhere in this study on the basis of atmospheric capture, peak deceleration, and vehicle lift and drag criteria and related

[redacted]

TABLE I-4-I ACCELERATION NOMENCLATURE

NASA: Vehicle Displacement		AMAL: Physiological Displacement		WADD: Equivalent Vehicle Displacement	Other	Unit
Symbol	Maneuver	Symbol	Description	Description	Description	
$+a_x$	Surge	$+G_x$	Eyeballs-in Chest-to-back	Backward facing	Transverse supine	g
$-a_x$	brakes on	$-G_x$	Eyeballs-out Back-to-chest	Forward facing	Transverse prone	g
$-a_z$	pull up	$+G_z$	Eyeballs-down Head-to-foot	Headward	Positive	g
$+a_z$	push over	$-G_z$	Eyeballs-up Foot-to-head	Footward	Negative	g
$+a_y$	side slip left	$+G_y$	Eyeballs-right Left-to-right	Rightward	Lateral	g
$-a_y$	side slip right	$-G_y$	Eyeballs-left Right-to-left	Leftward	Lateral	g
$\pm\dot{p}$	roll (right=+)	$\pm\dot{R}_x$	Roll			rad/sec^2
$\pm q$	pitch (up=+)	$\pm\dot{R}_y$	Pitch			rad/sec^2
$\pm r$	yaw (right=+)	$\pm\dot{R}_z$	Yaw (Left=+)			rad/sec^2

[redacted]

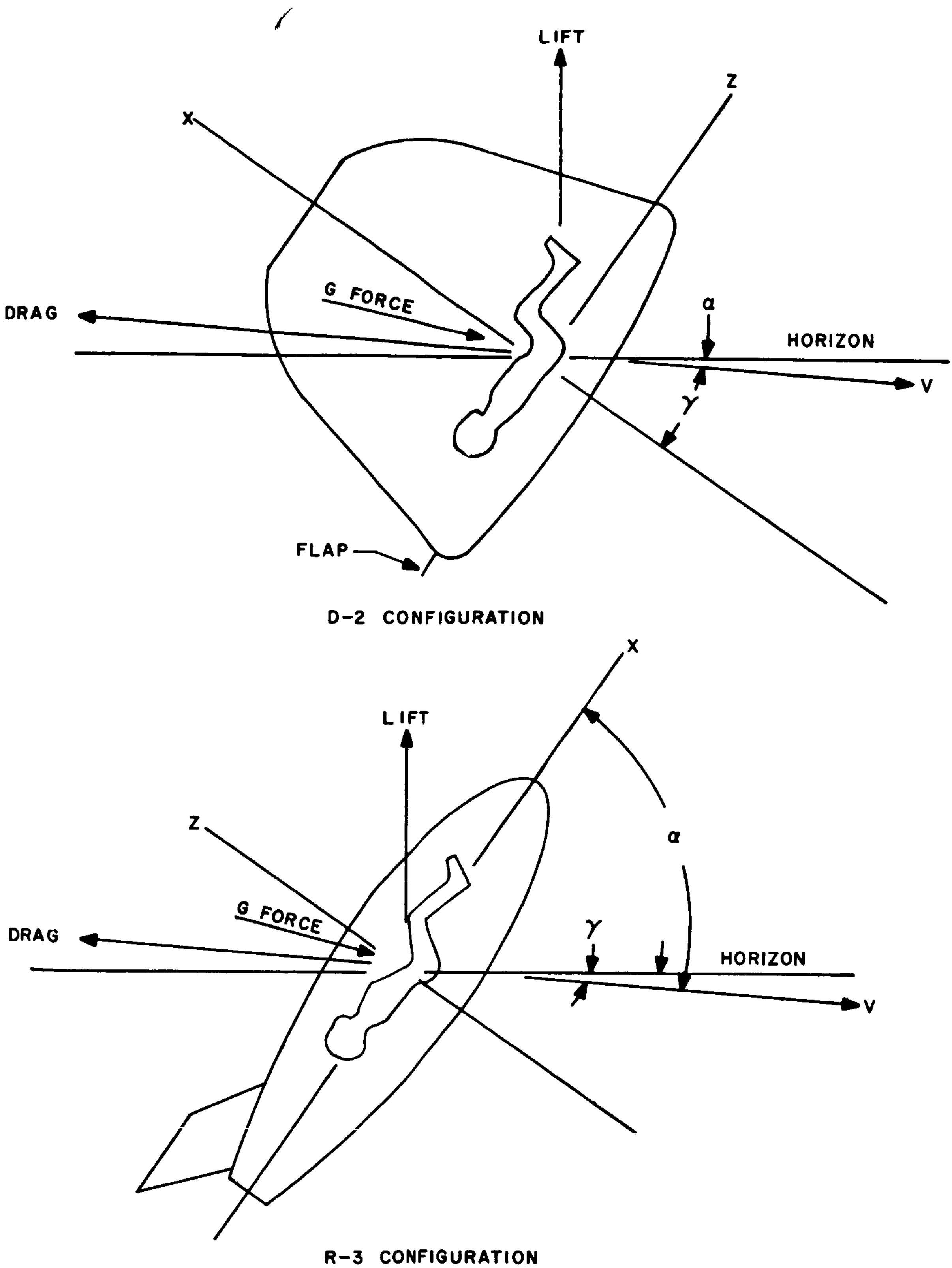


Figure I-4-1. D-2 and R-3 configurations

TABLE I-4-II ACCELERATION SYMBOLS

Symbol	Description
g	Acceleration of gravity (g units)
V	Velocity
Θ	Vehicle pitch angle
ϕ	Vehicle roll angle
α	Angle of attack (angle between longitudinal axis and velocity vector)
ϵ	Angle of attack error
X	Longitudinal body axis of vehicle
Z	Axis perpendicular to longitudinal axis in the plane of symmetry.
γ	Flight path angle
a_x	Acceleration along the x-axis of the vehicle
a_z	Acceleration along the z-axis of the vehicle

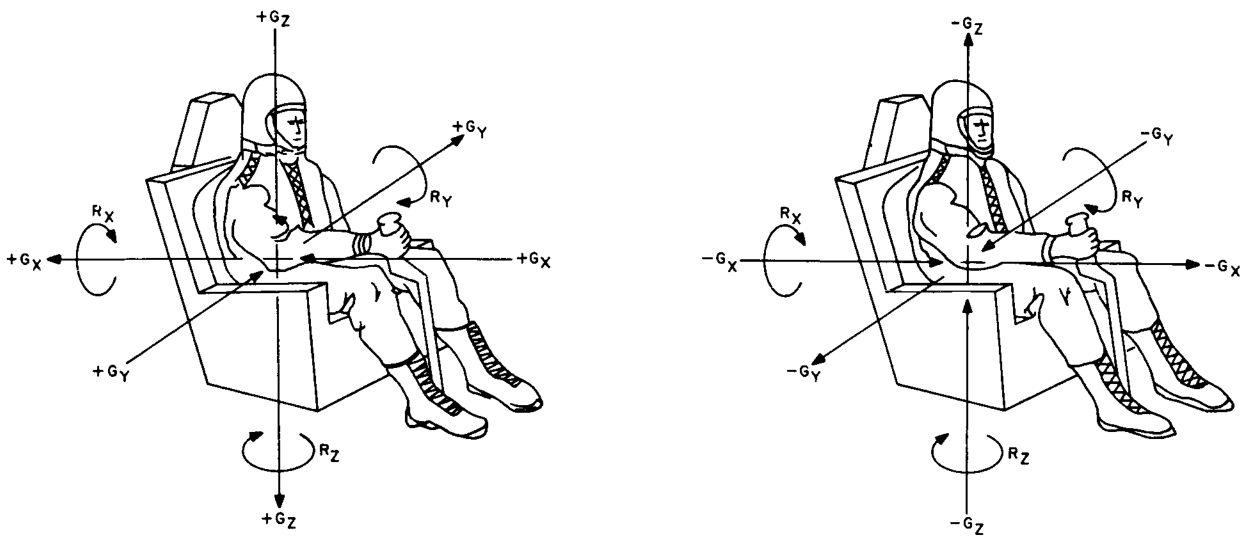


Figure I-4-2. Acceleration vectors

to such factors as dispersion, maneuverability, heat protection, and terminal guidance accuracy. Human acceleration tolerance is thus only one of a number of factors affecting the ultimate solution of the re-entry problem and can not be treated in isolation. Because utilization of aerodynamic lift for path control greatly alleviates the guidance accuracy requirement, pilot acceleration performance tolerance must be examined carefully in order to define his capabilities during re-entry deceleration as well as during terminal maneuvers.

A wide variety of sources of human physiological and performance tolerance data has been consulted for application to this problem. Several modes of presentation of tolerance data are in current use. Data are most frequently presented as G-time plots using a number of criteria such as voluntary tolerance (limited by pain or respiratory embarrassment), vision (greyout or blackout), or performance. Curves have been presented for a variety of body positions and the probable tolerance to intermediate vectors has been estimated by an elliptical approximation. An effort has been made to generalize tolerance data on the basis that the total energy dissipated during re-entry is given by the product of G and time. Hiatt has presented G x t versus peak G plots for several body positions. Such curves must be interpreted with care because the shape of the G-time curve has considerable significance, particularly when performance criteria are used. The total time is sometimes taken as the actual time at peak G or sometimes as that time above 90 percent of peak G (Smedal).

Continuous improvement in restraint systems has resulted in changes in accepted tolerance data. Smedal, for instance, gives equivalent performance tolerance data for both the + G_X and $-G_X$ positions, while Clarke and Hiatt show significant differences. At high G levels small changes in back angle and leg angle are important. Changes in back angle from 8 degrees to 12 degrees ($+G_X$ acceleration) noticeably increase visual problems because of the increased + G_Z vector. Variations in leg position will alter the amount of congestion and discomfort which occur.

Two types of performance tolerance studies have been conducted. Earlier studies involved open-loop operation of the centrifuge. Experimental tasks were performed while the centrifuge was driven in a prescribed G-time pattern. More recently, closed-loop studies have been conducted in which the subjects' control motions in response to an instrument display brings about appropriate accelerations. These are generated by computer circuitry so that the centrifuge performance simulates the aerodynamic control responses of the vehicle under study including appropriate responses of the flight instruments. As might be expected different performance tolerance values have been found under the two circumstances because kinesthetic feedback is present only in the closed-loop case. The data, however, may be affected by the presence of untoward angular accelerations which accompany centrifuge studies. Of the wide variety of performance criteria which might be used only a few have been systematically evaluated.

Tolerance studies also may be divided according to the type of vehicle under consideration including the restraint and support systems, G-time profiles and pilot tasks appropriate to its mission. Three types of such vehicle systems have been studied at Johnsville: the pure drag vehicle (Mercury), a semi-ballistic lifting vehicle (Eggers) and a winged vehicle (X-15). Studies have included only suborbital or orbital regimes, although some endurance tolerance data is available. Although considerable data have been generated, such data cannot easily be extrapolated or interpolated to new vehicle systems and flight regimes.

4.1.2.1 RE-ENTRY PHASE

Since it is considered mandatory that the crew of the APOLLO vehicle have the ability to fly the re-entry profile, the emphasis shifts from voluntary tolerance to that based upon performance. In other words, the re-entry corridor must be so shaped that during the most severe profile (steepest path angle) the acceleration level does not exceed the limits in either magnitude or duration that would prevent good manual control of the vehicle by the pilot.

Therefore, prior to discussing G limits, it is well to examine the re-entry maneuver and the task which the pilot will be expected to perform during this flight phase. Figure I-4-3 illustrates a typical velocity-altitude profile of a type of re-entry examined during the APOLLO study. The pullout phase of this type of profile provides the initial

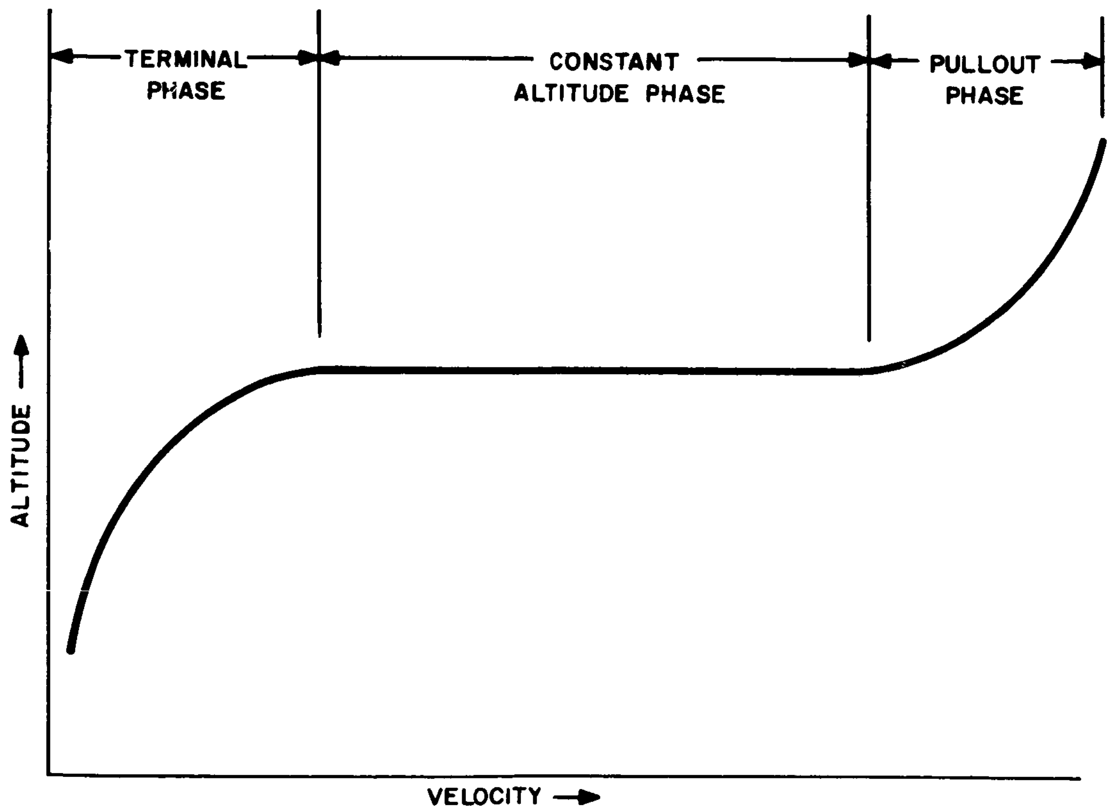


Figure I-4-3. Typical velocity-altitude profile vehicle deceleration and lasts until a constant altitude trajectory can be established. This entire phase is flown at a constant angle of attack. No range or cross range maneuvering is required at this time. Therefore, since the vehicle can be trimmed prior to re-entry at the angle of attack desired (-40 degrees), no control is required by the pilot at this time with the exception of maintaining this altitude. This function can be compared to a simple two-axis tracking task.

At the end of pullout, the vehicle will be nominally flying at greater than orbital velocity and will as a result require negative lift to maintain the desired altitude. Negative lift is achieved at a positive angle of attack. At this point, the pilot of the D-2 vehicle need only perform a simple 180 degree roll maneuver to effect this change in angle of attack, since the vehicle will rotate about an axis in the plane of the relative wind. The pilot of the R-3, however, must at this point pitch to zero angle of attack, then roll 180 degrees, and pitch again to the new angle of attack. These maneuvers, which are somewhat more involved than those during pullout, still require only two axis control at most with accuracies not critical with regard to the remainder of the re-entry profile.

Proceeding down the typical velocity-altitude profile within the constant altitude phase, the pilot now must modulate lift to zero as velocity decays and orbital conditions are achieved. At this point, the vehicles are again rolled 180 degrees to provide positive lift in an effort to maintain constant altitude as velocity decreases. It is during the constant altitude phase that both range and cross range maneuvering is accomplished. Roll and pitch control will therefore be required in response to information provided on the energy management display.

During the terminal phase, lift and drag will again be modulated through two axis control for final approach correction into the landing site. Vehicle control requirements terminate for the D-2 when the recovery parachutes are deployed except for orienting the vehicle under the parachute prior to impact. With regard to the R-3 final approach and landing control must, of course, be handled by the pilot. The only significance here however, with respect to the acceleration profile might be excessive fatigue induced by the rigors of this phase.

Acceleration-time profiles developed for the typical re-entry mode described above are shown on Figure I-4-4. This figure illustrates both the steepest (solid lines) and

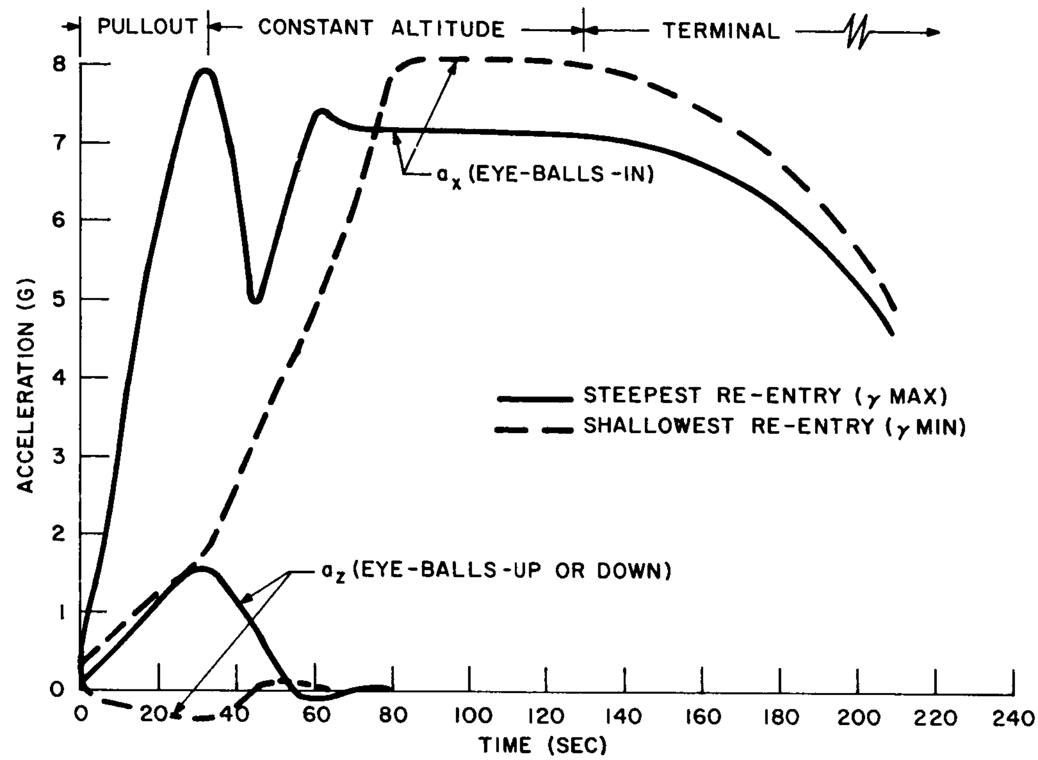


Figure I-4-4. Minimum range re-entry acceleration vs time

shallowest (dotted lines) allowable re-entry angles, and represents the minimum range program for both cases and therefore brackets the most severe acceleration conditions to which the crew might be subjected during this phase.

In examining these curves, we see that for the steepest re-entry case, G builds up at a rate of onset of approximately 1/4 g/sec to a peak of +8 G_X (eyeballs-in) and +1.5 G_Z (eyeballs-down) at the end of the pullout maneuver. The G-level now decreases with dynamic pressure until lift is modulated to zero at orbital velocity (maximum drag configuration). In the case shown, drag is maintained at a high level, since for minimum range it is desirable to reduce velocity as rapidly as possible. During the terminal phase, G falls off rapidly with velocity and altitude until terminal velocity is reached just prior to parachute deployment.

Recent studies of Smedal indicate that the above profiles fall within man's tolerance limits. If peak G and time at 90 percent of peak G are taken from the steep trajectory and applied to Smedal's curve, Figure I-4-5, we see that this trajectory produces tolerable G loadings. Furthermore, current work by Smedal (which is continuing at

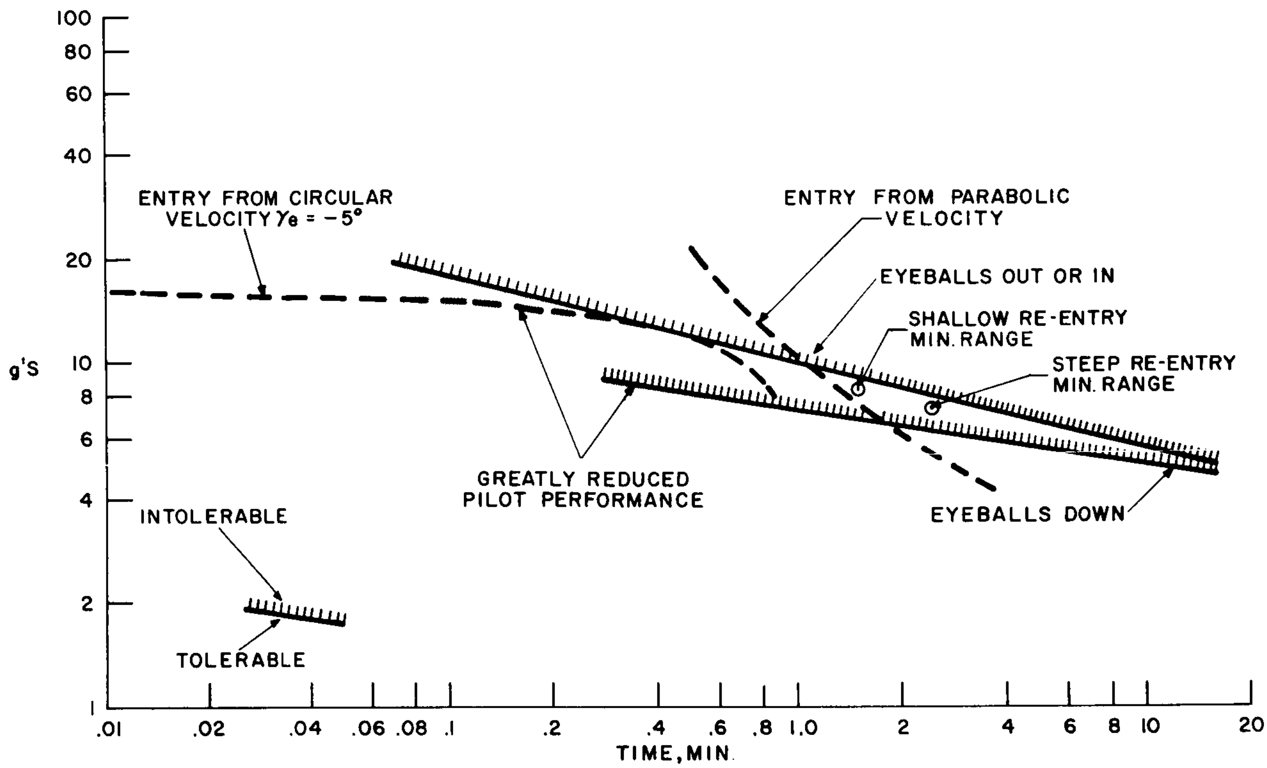


Figure I-4-5. G-time tolerance (after Smedal)

this writing) would indicate that performance of relatively complex five degree of freedom tracking tasks can be maintained at this or higher G levels. Preliminary results of this work suggest that the tolerance limits shown in Figure I-4-5 are conservative. Vehicle oscillation and the inclusion of man in the control loop with the possibility of overcontrol would suggest that the re-entry angle limits as described by the illustrate profile are reasonable for this mission.

The second pair of curves illustrated in Figure I-4-4 representing the shallowest re-entry minimum range condition show a slower rate of onset and a lower G-level at pullout and a resultant higher G-level during the constant altitude phase. Here peak G may be taken as 8.2 and time at 90 percent of peak as 87 seconds. Plotting this on Figure I-4-5, we again see that tolerance limits are not exceeded and that performance will be maintained at a level sufficiently high to satisfy the requirements of these vehicles.

The role of the pilot during re-entry and his control-display system is discussed in Chapter II Section 2.0 of this volume and the general physiological and psychological effects of acceleration are discussed in the Appendix.

4.1.2.2 LAUNCH PHASE

No new acceleration problems are expected for Saturn launches. A typical acceleration-time profile is shown on Figure I-4-6. Although propellant burning times are longer, the maximum G-levels are not much greater and the G-time curves (as presently known) are within human tolerances. Such curves will be simulated on the centrifuge for detailed analysis during the hardware phase. Problems to be studied include (1) respiratory functions, which might be limiting because of the long exposure times (770 sec), and (2) the combined effects of vibration and acceleration.

The acceleration transients induced by booster cut-off must be reconsidered for the Saturn case when human tolerance data have been accumulated on Mercury flights. No adverse effects have been demonstrated in animal flights to date. In addition, theoretical

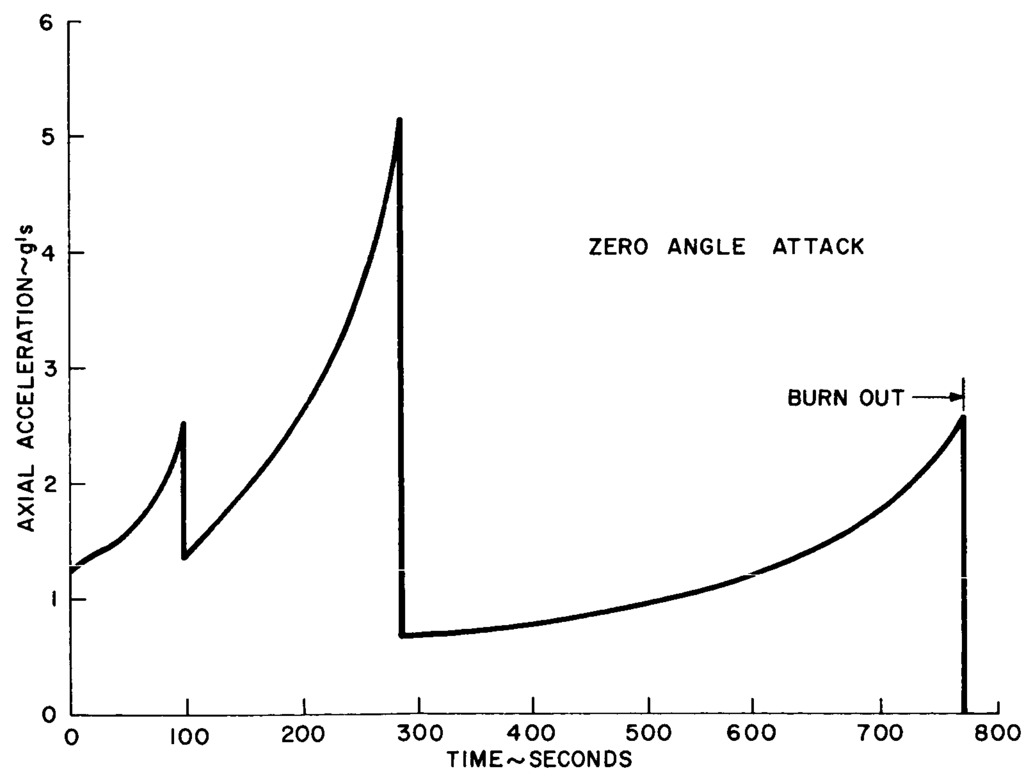


Figure I-4-6. Axial accelerations for a nominal APOLLO ascent trajectory considerations do not suggest problems. For instance, if man is considered a spring-mass-dashpot system which is stressed by acceleration forces, the release of such a system to zero g should result in oscillations at the system's natural frequency. Man is highly damped, but consideration must be given to the natural frequency of the man as well as his seating and restraint system in order that excessive oscillation does not occur.

Consideration has been given to the role of man during the powered flight portion of the mission. For reasons discussed elsewhere, his functions during launch consist primarily of monitoring the performance of the vehicle and boosters. He obviously has the capability to initiate abort or to select a secondary mission should conditions require. These functions are simply performed and will easily be accomplished within the launch acceleration profile.

4.1.2.3 ABORT PHASE

Launch pad abort represents a serious problem for Project APOLLO. A major system consideration is the protection of the pilot in the event of a Saturn detonation. A detailed consideration of this occurrence (Chapter II Section 3.0) has resulted in a requirement to propel the man to a distance of 524 feet in 1.78 seconds in order to maintain the over-pressures within reasonable limits (10 psi).

In order to accomplish such an escape, high G-loadings may be expected. Figure I-4-7 indicates the extremes which will occur. Escape rockets required for this separation will produce a peak of approximately 15 G_X (eyeballs-in), tapering to about 7 G_X in 2.5 seconds. An addition G-vector of as much as 4 g may be present depending on the altitude of the vehicle and the occupant. This vector may be present in the y or the z-axis or may fluctuate. At burnout, aerodynamic drag will result in the rapid reversal of the G_X vector to approximately -10 G_X (eyeballs-out) with a less rapid decay toward zero. Separation of the re-entry vehicle and exposure to a larger drag surface results in the reapplication of the $-G_X$ acceleration. Parachute opening shocks are not shown but are low during this phase.

Experimental data developed on the Johnsville centrifuge have indicated that the rapid variation in the G vector from + 9 G_X to -9 G_X in one second is tolerable. Variation

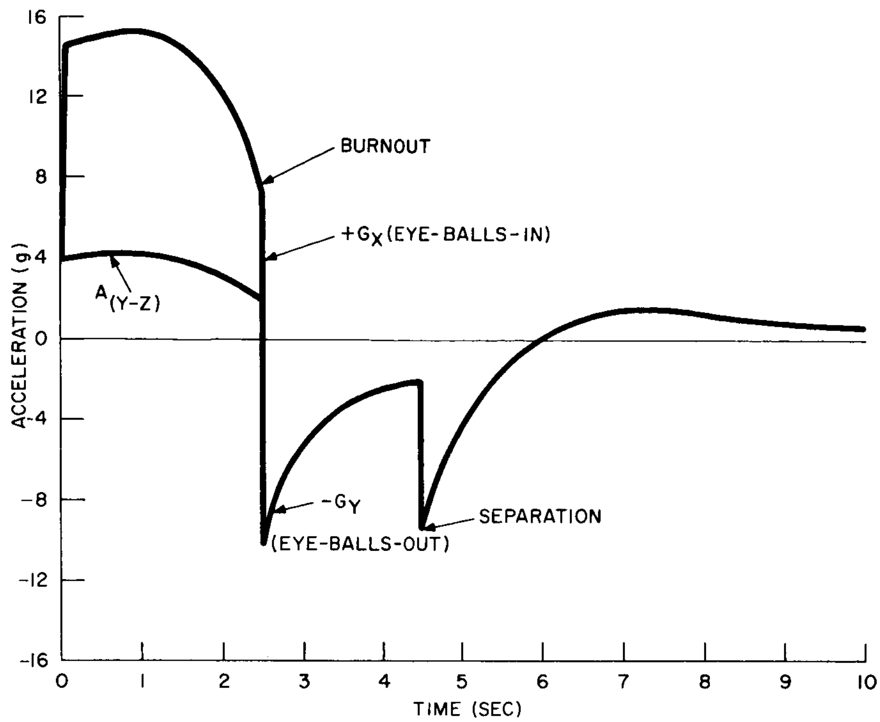


Figure I-4-7. Launch pad abort acceleration-time profile

from +11 G_X to -11 G_X is marginal. If the abort criterion is the G-level value developed at Johnsville, i.e., ± 10 g, risk of overpressure damage is incurred. Calculations of this overpressure value are presently made on the basis of a number of assumptions used to characterize a Saturn explosion and may be open to criticism. On the other hand, design limits of 20 G_X and 10 G_{Z-y} have been suggested in some NASA documents.

Because of aerodynamic consideration (low drag shape) lower G values will generally be experienced for the R-3 vehicle. Escape in this vehicle requires immediate performance by the pilot who must effect a near-conventional landing. The landing sequence of the semi-ballistic vehicle (D-2) proceeds automatically, except that in the event of a failure of parachute deployment a manual override control must be activated. If the lenticular vehicle (R-3) is recovered by parachute following escape, its shape and the presence of wind and thus drift may result in undesirable tipping moments at impact. Although the crew of the semi-ballistic vehicle are positioned so as to experience launch, re-entry, escape rockets, parachute and impact acceleration loadings in the

$+G_X$ direction, parachute and impact loadings for the lenticular vehicle must be taken "on the straps" in the present configuration.

It is apparent that in either case considerable attention must be paid to the seating and restraint system. Such a system must not only provide protection against the simple $+G_X$ or $-G_X$ vector but must support the man in all directions and under a variety of loading conditions. The fundamental requirements to be met are not only the coupling of the man to the vehicle or his seat system but the maintenance of the proper coupling among the various parts of the body. Of particular significance is the coupling of the extremities to the torso and the coupling of the head to the shoulder girdle. In general, more damage may be done by differential displacement of the various parts of the body than by any other mechanism. While elaborate massive restraint systems may successfully accomplish this objective, such systems may be difficult to assemble and don and difficult to remove within the vehicle. The recent development of a restraint system by Smedal offers promise and is described in detail elsewhere. Preliminary data generated on the Johnsville centrifuge are encouraging, but the system has yet to be tested under escape or impact loads.

The launch pad abort represents the most severe case with respect to the acceleration environment. Abort at maximum dynamic pressure while causing a $-G_X$ acceleration similar to the launch pad case does not apply as high a $+G_X$ vector during separation from the booster. As a result the overall change in G is lower and therefore considered less severe.

4.1.2.4 LANDING PHASE

The method proposed for impact attenuation of the APOLLO vehicle is discussed elsewhere in this report. Considering the worst design condition of impact with only two of the three chutes deployed and resulting impact velocity of 36.8 ft/sec, an acceleration-time curve as illustrated on Figure I-4-8 can be expected. Examination of this curve indicates a rate of onset of 400 G/sec and a peak of 14.0 G_X (eyeballs-in).

Superimposed upon this curve are spineward acceleration tolerance data from Goldman and Von Gierke. From this plot it can be concluded that this emergency case impact will not exceed human tolerance presuming that an adequate seating and restraint system is provided which will prevent undue displacement or rebound.

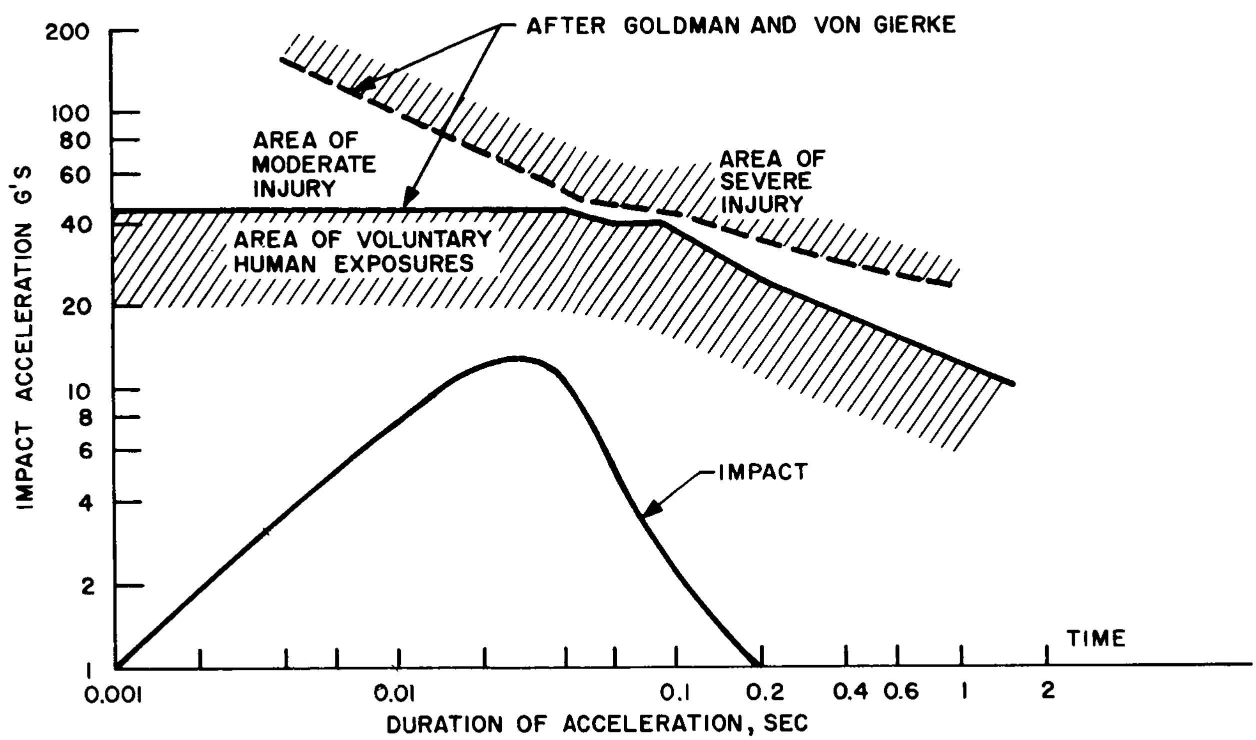


Figure I-4-8. Acceleration vs time curve of two parachute impact

Another factor which must be considered is the possibility that the vehicle, upon impacting on water, may "tuck under" or upon impacting on land, may tip over violently and result in accelerations in any direction. While the work accomplished for the landing systems for this vehicle indicates that this is an unlikely situation, the possibility does, however, dictate the inclusion of a restraint system which is multidirectional. Included for reference purposes are the tolerance curves from Goldman and Von Gierke for sternward, tailward and headward accelerations (Figures I-4-9, I-4-10 and I-4-11).

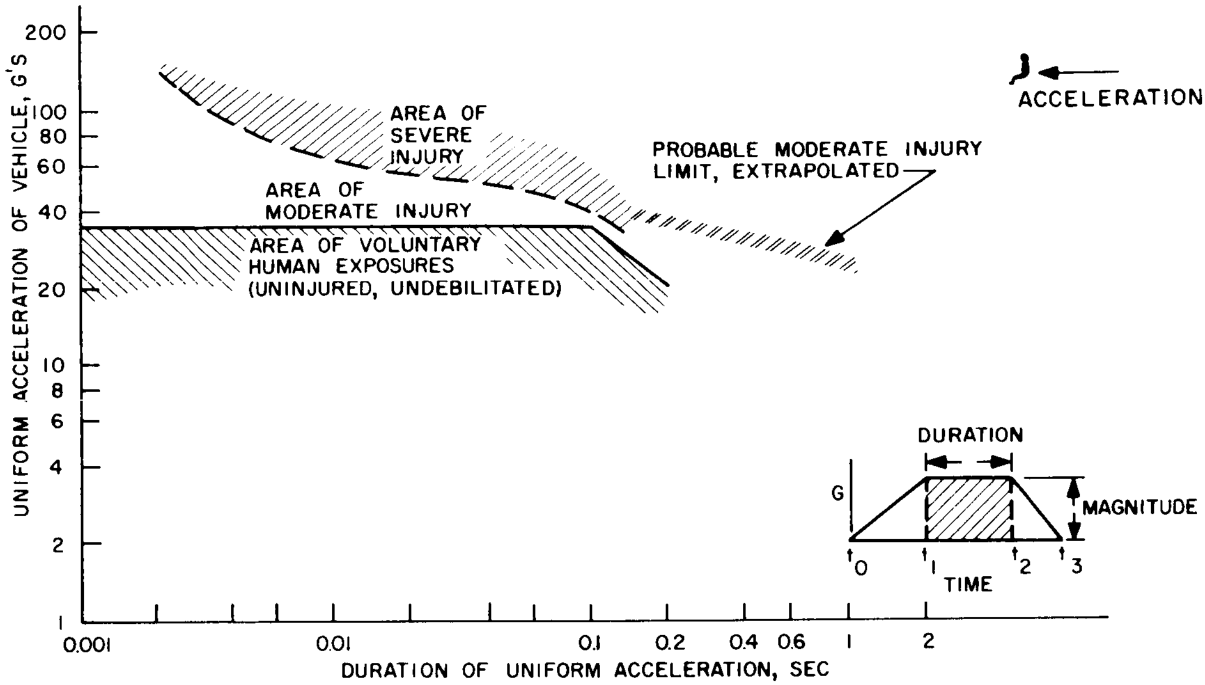


Figure I-4-9a.

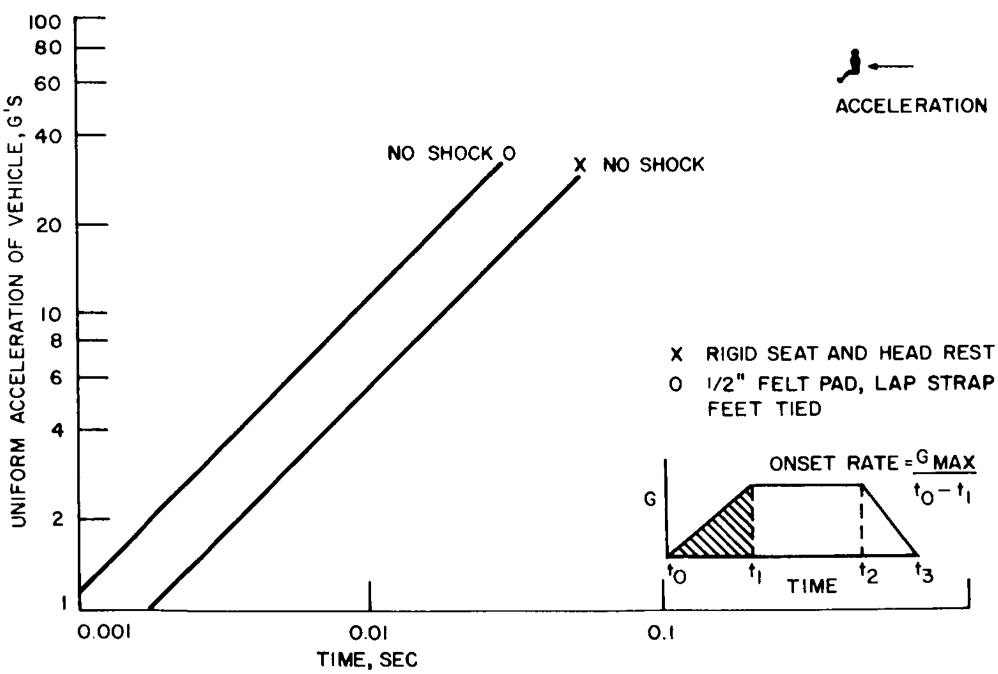


Figure I-4-9b.

Effect of rate of onset on sternumward acceleration tolerance (after Goldman and VonGierke)

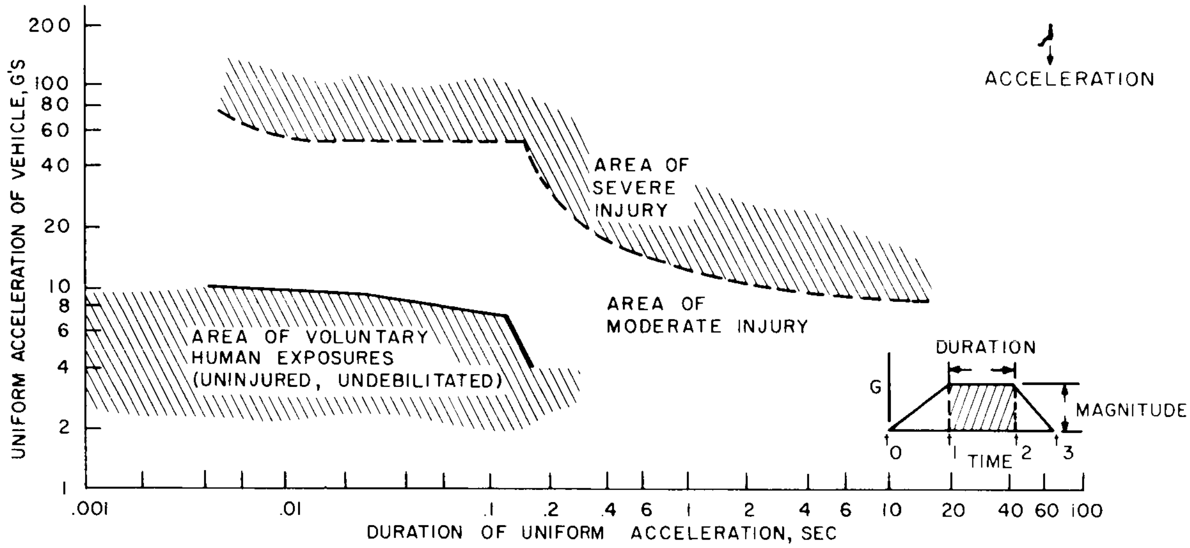


Figure I-4-10a. Tolerance to tailward acceleration as a function of magnitude and duration of impulse (after Goldman and VonGierke)

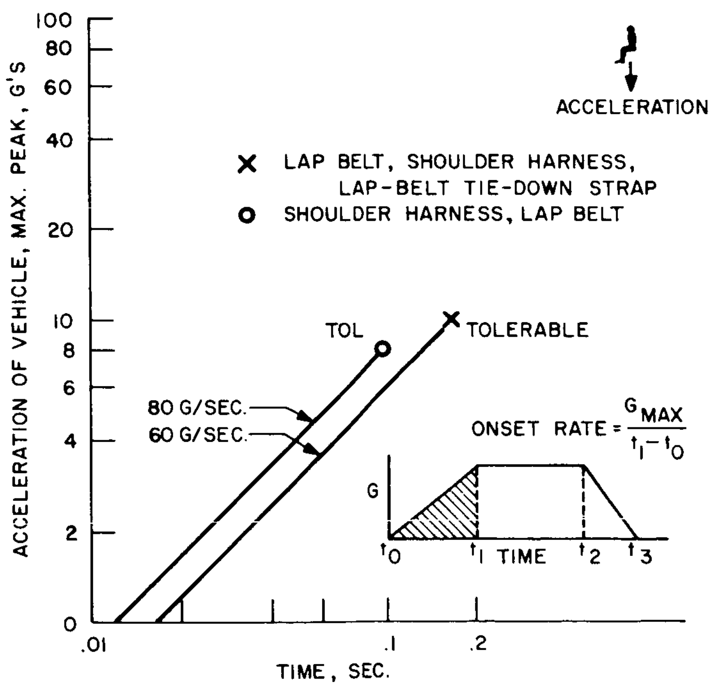


Figure I-4-10b. Effect of rate of onset on tailward acceleration tolerance (after Goldman and VonGierke)

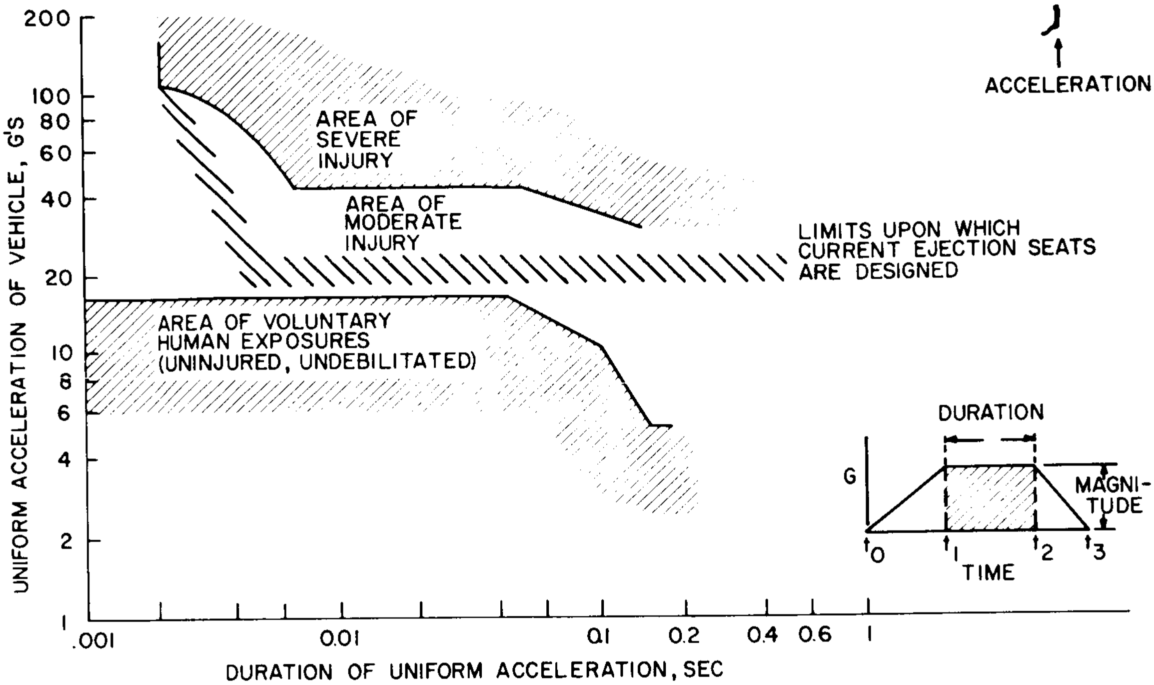


Figure I-4-11a. Tolerance to headward acceleration as a function of magnitude and duration of impulse (after Goldman and VonGierke)

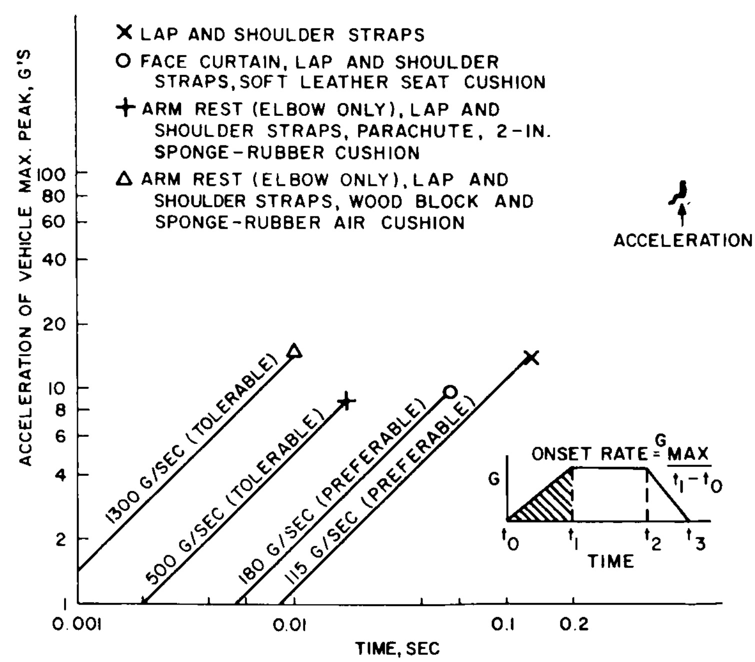


Figure I-4-11b. Effect of rate of onset on headward acceleration tolerance (after Goldman and VonGierke)

4.1.2.5 SUMMARY

In summation, a systems approach has been used in developing the acceleration profiles just reviewed. Each phase of flight has been analyzed not only with respect to man's physiological tolerance to acceleration, but also with regard to his performance requirements during any portion of any phase.

As an example, during re-entry when a substantial performance requirement is superimposed upon the requirement for survival, the acceleration limits have been shifted down toward lower G levels (8-8.5 G maximum) in order to permit the crew to control the vehicle within the limits required with respect to energy management. During abort however, when no control is required until after acceleration decay, the G levels have been shifted upward closer to the physiological limits yet not beyond a point which will prevent post acceleration performance as required. In this manner, considering man as a variable function component, acceleration-time profiles have been evolved which minimize the compromise to both vehicle and crew.

Much is yet to be learned about man's performance in an acceleration environment and the limits established above have in some cases a relatively large effect upon the design of the vehicle and vehicle subsystems. Therefore, within the first six months of the hardware program, centrifuge studies will be conducted wherein Saturn launch and lunar re-entry profiles will be simulated. Performance will be measured using tasks, controls and displays identical to that required for the APOLLO mission. In addition, tolerance to abort loading and post acceleration performance will be measured. While a centrifuge cannot completely duplicate the abort profile, live tests on this apparatus possibly coupled with sled tests which measure dummy displacement should satisfactorily demonstrate both performance and tolerance capability.

Later in the program as restraint system, cocoon, display and control hardware becomes finalized these profiles will again be simulated in order to substantiate the performance and survivability of the crew within the acceleration environment of the APOLLO mission.

4.1.3 Seating and Restraint

During the early portion of this program an investigation and analysis was made of several types of seats which might be suitable for the APOLLO vehicle. Those seating concepts studied included moulded couch, contoured foam, micro balloon and raschel net. Based upon mission requirements assumed, weight and flexibility, the raschel net seat was tentatively chosen for use in this vehicle.

Continuing study of these mission requirements, packaging, integration with the cocoon and additional analysis of other concepts has led to the final selection of a system other than the raschel net seat. The system selected for APOLLO is one which was designed and developed under the direction of Captain H.A. Smedal of the Ames Research Center, Moffet Field, California. This system which is, at this writing, undergoing tests at the Aviation Medical Acceleration Laboratory, Johnsville, Pa. appears to be the most sophisticated and most highly successful concept of seating and restraint yet proposed. Preliminary results of this work indicate, as mentioned above, that G-tolerance may well be expected to exceed these established earlier by Smedal (Figure I-4-5).

The system, illustrated in Figures I-4-12 and I-4-13, was designed specifically for use in tne centrifuge with a definite eye toward reduction of machine down-time. As a result it is at the present not suitable for installation in a vehicle. However, the concept and general hardware approach of Smedal is directly applicable to vehicle use. Different mounting arrangement, seat-pan adjustment and changes to the harness to permit ease of hook-up and quick release are in essence all that would be required to adapt this system to APOLLO.

Specifically, the seat back which is roughly contoured around the torso will be rigidly mounded to vehicle structure. A seat pan will replace the thigh supports shown on Figure I-4-13 and will be adjustable in flight to accommodate the differences in shoulder to buttock height of various crew members. In essence the short man raises the pan until his shoulders are in the correct position with respect to the seat back. This incidentallly places his eye within ±0.15 inch of optimum. The tall man, on the other hand, lowers the seat pan to suit his longer torso, again indexing his shoulder to the seat back. Since eye to shoulder distance varies only 0.3 inch from the 5th to 95th percentile sizes the tall man will also be within 0.15 inch of optimum eye point.

Figure I-4-12. Front view - restraint system after smedal

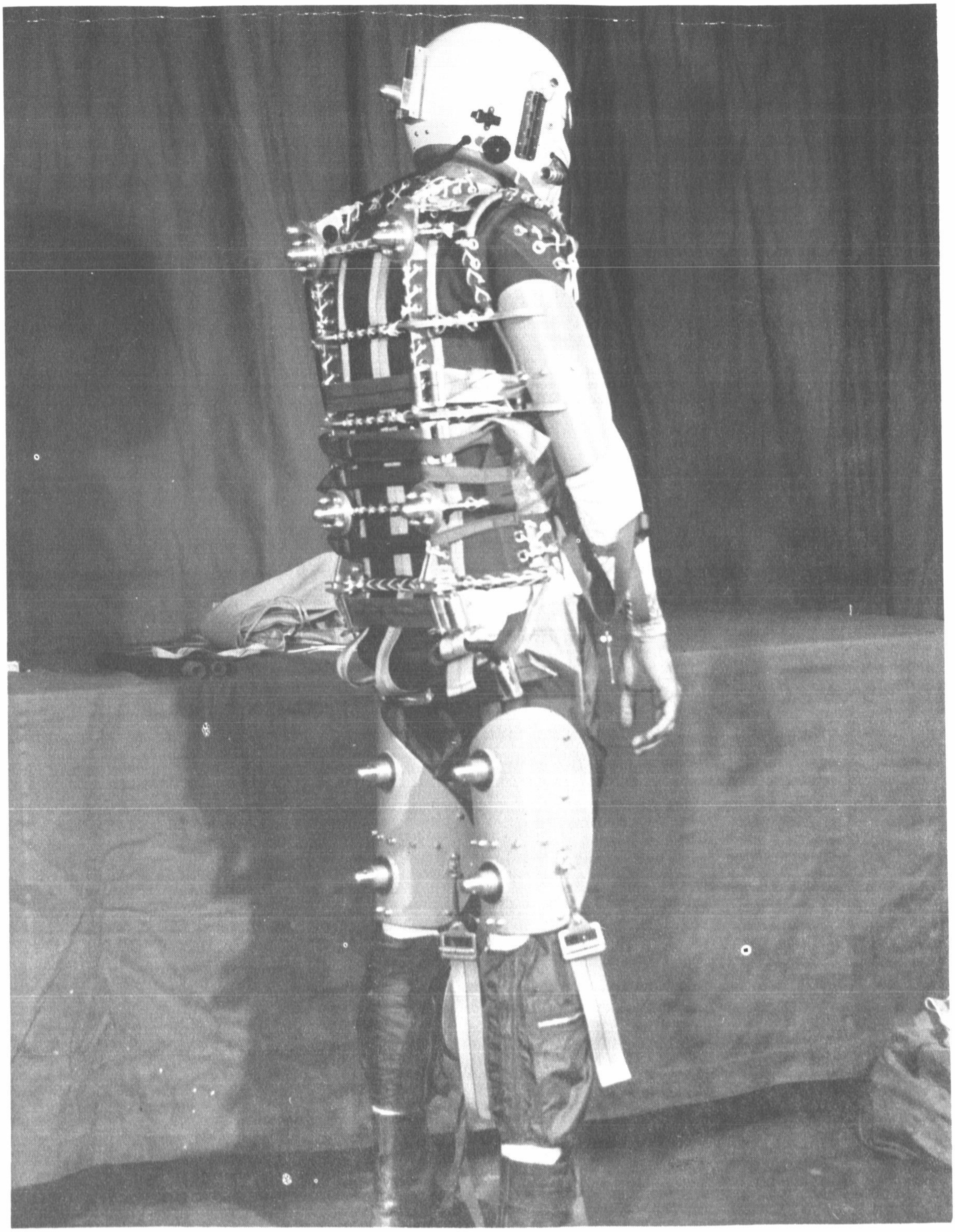


Figure I-4-13. Rear view - restraint system after smedal

The upper torso is restrained through a vest which is secured to the seat. Modification involves design for ease of adjustment, four point support and quick release. A bladder placed between the seat and the back of the occupant will, when inflated, effectively tighten the harness and also provide some measure of total body support.

Head restraint will be provided in a manner identical to Smedal. The occupant will wear a helmet with a molded face piece. The helmet is secured to the structure through one connection at the rear which permits vertical displacement of the head while still preventing fore, aft or rotational movement.

The entire structure of the seat back, headrest, arm rest and arm supports integrates directly with the cocoon structure.

While exhibiting excellent characteristics during re-entry accelerations, this system has yet to be tested for impact and sudden G-reversals characterized by abort. However, the system has demonstrated performance capabilities in the eyeballs-out direction equal to that for the eyeballs-in direction. This would indicate more complete restraint than ever before achieved and certainly a high degree of multidirectional capability.

It can therefore be assumed that this system, modified for vehicular use, can be made to perform throughout the APOLLO acceleration environment.

The crew within the D-2 configuration will be oriented with the plane of the back perpendicular to the "X" or longitudinal axis of the vehicle. In this position they will receive launch, re-entry, parachute opening and impact accelerations in the most favorable $+G_x$ direction. Abort G is also taken in this direction with drag-induced G at abort rocket burnout received in the $-G_x$ direction.

The crew of the R-3 are also oriented with the plane of the back perpendicular of the longitudinal axis of the vehicle for launch, abort and landing. During re-entry, however, the seats are rotated approximately 90 degrees so that the crew are once again most favorably positioned with respect to the acceleration vector (see Figure I-4-1). Instrumentation and controls will be rotated with the seat. This is, incidentally, another distinct advantage of the seat system chosen, in that by nature, this rotation can be accomplished without additional complication of the seat hardware, per se.

Again referencing Figure I-4-4, it is seen that during the steepest re-entry condition (maximum path angle) $-a_x$ peaks at only 1.5 G. This low level of eyeballs-down acceleration obviously does not require the use of an anti-G suit as an auxiliary measure of protection. The seating and restraint system described above, however, can integrate an anti-G suit within it should mission requirements generated later in the program create a situation where a device of this nature would be of value.

4.2 SECONDARY PRESSURE PROTECTION SYSTEM

4.2.1 Systems Analysis

A secondary pressure protection system must provide an emergency atmosphere when either cabin pressure integrity is lost or the cabin environment becomes irreparably contaminated. Cabin pressure integrity can be lost through structural failure caused by any one or a combination of loading, vibration, shock or meteoroid impact. A marked increase in design leak rate from any cause is also considered pressure vessel failure. The cabin atmosphere can, of course, be contaminated by the products of fire, of-fire control, radiation, materials out-gassing and arcing electrical equipment.

For the sake of simplicity it would be well to discuss this latter area first in order to eliminate it from further detail consideration. This elimination is possible merely by pointing out that the build-up of toxic gases within the cabin will be at a relatively slow rate and hence will have little effect on the design or selection of a secondary-pressurization system. That is to say, regardless of the method of achieving a new atmosphere, the ''time of action'' requirement for protection against contamination can satisfactorily be accomplished. As an extreme case the cabin could be purged of these gases at no total pressure reduction. Although this is an inefficient method from the standpoint of stored gas used, it illustrates that cabin contamination protection will not be a governing criterion for the secondary pressure protection system.

One of the most important criterion, however, in the design of this system, is the time available to take corrective action during a decompression. This time is defined herein as the interval during which the oxygen partial pressure drops from the nominal 180 mm Hg to the minimum required to prevent the onset of hypoxia.

From above it is seen that this decompression can occur either through structural equipment failure or meteoroid puncture. Considering these in order, it appears most reasonable to assume that structural failure, if it occurs at all, will occur during the launch or re-entry phases of flight, since these are the periods of maximum loading, vibration and shock. It is therefore assumed that the pressurization system must be such that it can be at "ready" during both of these phases. In the case of the pressure suit, as an example, the crew would be wearing the garment hooked up with face plate closed, during these periods.

Having satisfactorily survived launch, the cabin must be assumed to be structurally sound throughout orbital flight. While it is conceivable that leaks may occur or increase in size during this portion of the mission, the elimination of virtually all but pressure loading from the cabin structure permits an extremely high level of confidence to exist with regard to pressure integrity.

We are therefore left as a design consideration only that decompression which might be caused by meteoroid impact. A discussion and analysis of meteoroid impact and pressure vessel penetration is presented elsewhere. This analysis indicates that a meteoroid large enough to cause a hole of 1-inch in diameter will penetrate the mission module once every two hundred missions. Similarly, a penetrating meteoroid will cause a one-inch hole in the command module once every six hundred forty missions. Entering the curve of decompression time versus hole size shown on Figure I-4-14 with this one-inch diameter hole, we see that the time for the oxygen partial pressure to reach 90 mm Hg is 51 seconds for the D-2 and 54 seconds for the R-3 configuration. If we add to these numbers one minute of useful consciousness, we see that roughly two minutes are available to take corrective action.

It is now necessary to discuss another design criterion which imposes constraints upon the secondary-pressure protection system. This criterion is that one which dictates the requirement for "shirt-sleeve" flying capability. As applied to the secondary-pressure protection problem, it simply indicates that the system must be designed so that the crew need not wear a pressure suit nor any other protective garment continuously. Considering the "time of action" arrived at above, a limit is presented on the use of the pressure suit, since it is inconceivable that this device could be donned during the time available.

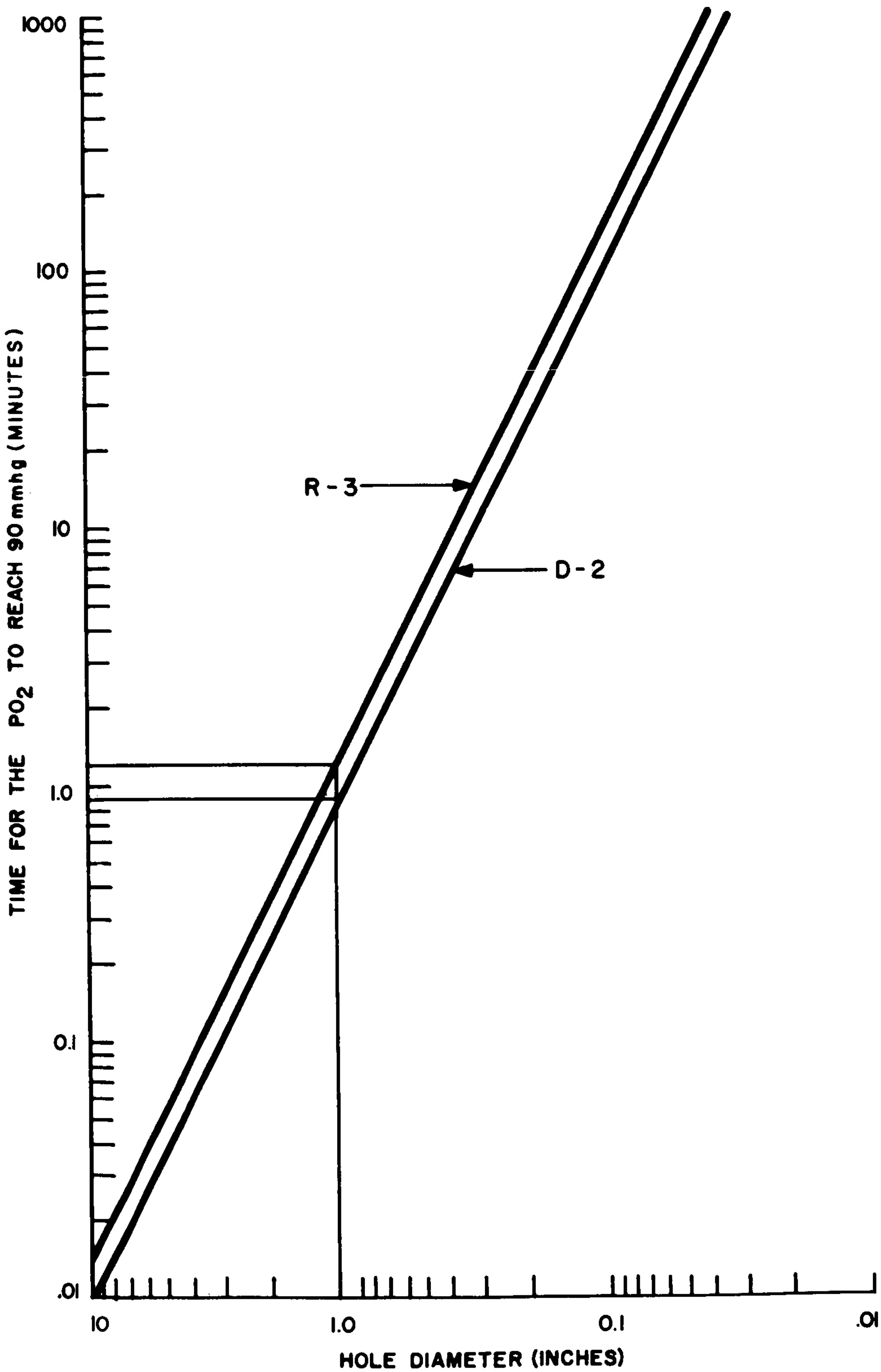


Figure I-4-14. Decompression time versus hole size

Use of the pressure suit, however, cannot be completely ruled out, since the vehicle is designed with two compartments and since the ''shirt-sleeve'' philosophy can be modified to permit one member of the crew to be clothed at all times. Under these conditions the other two members would, at penetration of the re-entry vehicle retreat into the mission module and seal the hatch behind them. The suited crew member then attempts to repair the leak, and can, upon success, repressurize the compartment. Should the puncture be irreparable, then the other two crewmen can don their suits under no particular time constraint, enter the re-entry vehicle and make preparations for return. In short, it is evident that even though our calculated "time of action" is on the order of two minutes, the use of two compartments permits the pressure suit to satisfy the criteria established thus far. It is pointed out, however, that an airlock must be provided in order to reduce the weight of atmosphere lost during transfer from one compartment to the other.

It is now appropriate to discuss the second method of emergency pressure protection to be analyzed. This method employs an encapsulated seat principle similar in configuration to the escape capsules used in the B-58 aircraft and scheduled for the B-70. These ''cocoons'' will perform the same basic function as the suit, but can by nature be closed rapidly so that the time to get to a seat is virtually all the time that is necessary to become protected.

This means that, upon decompression, the crew, two members of which will normally be in seats, will simply close the hood and remain inside during the course of the emergency. Now, since the requirement to attempt repair exists, one suit must be provided within at least one cocoon. This crewman now dons this suit and exits into the decompressed vehicle to perform the repair if possible. If repair cannot be made, then he retreats back into the ''cocoon'' and return is initiated. The primary vehicle controls will be arranged such that they are located within the cocoon enclosure for use during this emergency.

Since the suit has limited use, it need not be as sophisticated as those normally considered for use in vehicles of this nature. It may, in fact, be optimized for the job to be performed, and ease of donning with leakage and long-term habitation requirements minimized. Consultation with leading pressure suit manufacturers has resulted in the overwhelming opinion that a suit of this nature can be constructed within the present state-of-the-art. Packaging will be optimized through the use of a soft helmet and adjustment will be provided such that one suit will accommodate the entire crew.

Having considered only the "time of action" constraint, it would appear that both the suit and "cocoon" approaches are suitable for this vehicle. However, other criteria must be considered prior to making a selection. One of primary importance is that of habitability, for except in emergencies occurring at the extreme beginning or end of the mission, the time to return can be as long as 3 to 4 days. Unfortunately, little testing of long-term pressure-suit habitability has been accomplished to date. However, the data which does exist is rather discouraging for confinement times of 3 to 4 days. Briefly, the conclusions one might reach after examination of this information is that survival is probable, but performance can be expected to be severely reduced.

Body waste handling and feeding for this relatively long period presents certain problems. While the urine and feces containment problem could conceivably be ignored, closed face-mask feeding of the bite-sized food form which is chosen for this mission presents a problem not easily solved.

The use of the cocoon, however, suggests that (1) habitability problems and performance decrement will be substantially reduced and (2) feeding and waste handling will present no unsolvable design constraint.

Another area yet to be discussed is mission completion possibilities when design leakage is substantially exceeded as a result of the rigors of launch. Should this situation occur, then the low habitability of the pressure suit would render abort mandatory. The use of the cocoon, however, may permit mission completion by retreat of the crew into these units, sealing and remaining there until the vehicle has reached the proximity of the Moon. Having in this manner conserved the on-board oxygen and nitrogen supplies, the cabin could be repressurized and the crew could emerge from the cocoons to perform the tasks necessary during lunar orbit. Re-entering the cocoons, the crew now can return by once again reducing the effective leakage loss.

Mission flexibility is a very basic and important design criterion for equipment under consideration for this program. The modular construction of the vehicle strongly implies the use of the re-entry vehicle for any mission without necessarily combining it with a mission module. Since under this flight condition the "shirt-sleeve philosophy" will still exist and since the "time of action" will, as a result of using only one compartment, be reduced, the pressure suit must be considered unsatisfactory. Therefore, in order to preserve the mission flexibility requirement, it will be necessary to incorporate in the re-entry vehicle an emergency pressurization system other than a pressure suit. The encapsulated seat is therefore proposed.

4.2.2 Cocoon Design

Photographs of a "cocoon" mock-up are presented as Figures I-4-15 and I-4-16. Shown also are the seat back and seat pan integrated with the cocoon structure. This integration provides a total seat/restraint/cocoon system weight equal to or less than that anticipated for a seat/restraint/pressure suit combination.

The hood of the cocoon will be fabricated of neoprene-coated nylon with a plexiglass segment inserted to provide a transparent area of high optical property. This hood will be closed by pneumatic cylinders operating from the high-pressure nitrogen supply. This permits the hoods to be closed during launch and re-entry when acceleration loadings will prevent manual operation.

Critical displays and controls are incorporated within the enclosed section of the center (vehicle commanders) cocoon so that all functions necessary for midcourse flight and re-entry may be performed. Seals will be provided around the mating surface. These seals will be either pneumatic, gasket or a combination of both. Each cocoon will be sealed to a leakage rate of 300 cc/min or less. This offers a total leakage for three cocoons slightly less than that assumed for the vehicle. Therefore no additional oxygen is required to compensate for the possibility of living within the cocoon for extended periods. This leak rate is considered extremely realistic when it is compared with the rates achieved with pressure suits which have greater sealing lengths and more difficult sealing tasks. Also, the availability of the seal area to the occupant can be used to advantage to assure low leakage by manual application of additional sealing materials.

Figure I-4-15. Cocoon mock-up - front view

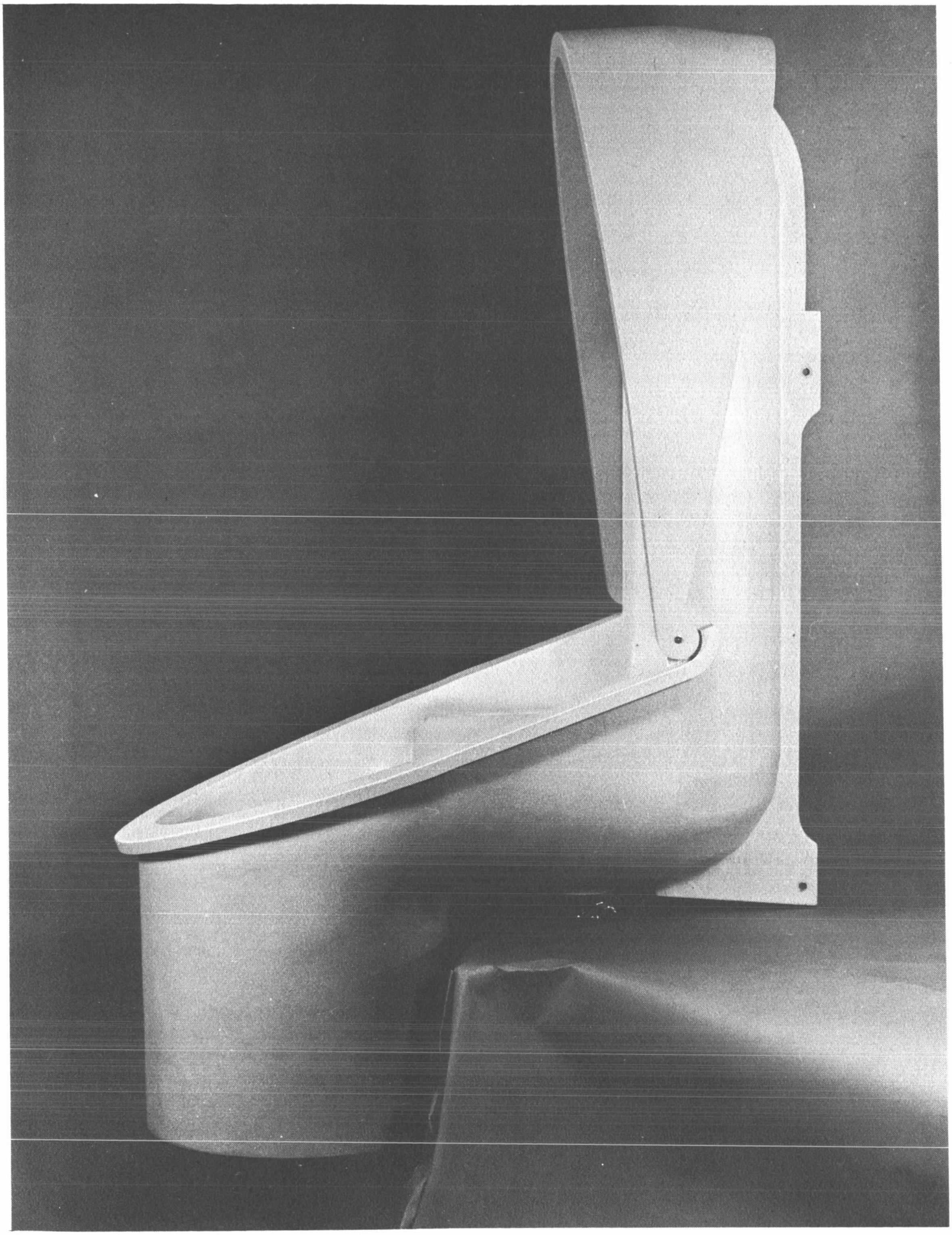


Figure I-4-16. Cocoon mock-up - side view

4.3 ACOUSTICAL ENVIRONMENT

To determine the magnitude of the noise level to which the crew of this vehicle will be subjected, an analysis was performed which attempts to predict the internal acoustical environment. Of primary interest was investigation of the need for the inclusion of supplementry attenuation material in the cabin structure and also the constraints which might be placed upon the communication system as a result of this noise level.

The primary sources of acoustic noise which affect any re-entry vehicle may be divided into three categories as follows:

1. Rocket engines during the static firing and launching of the missile.
2. Turbulent aerodynamic boundary layer during the boost phase of the vehicle.
3. Re-entry turbulent aerodynamic boundary layer and wake of the re-entry body.

The flight phase following boost may be briefly mentioned for the sake of completeness of the trajectory picture. The internal noise level during this portion of the flight will be low, governed only by the noise level of the on-board equipment. This takes the reverberation characteristics of the space cabin into consideration. For example, it might be assumed that the ambient noise level of the on-board air conditioning and other sound-producing systems is equal in magnitude to that of the Mercury Space Capsule air conditioning system, i.e., 87 db. For comparison, it would correspond to that sound level of a loud conversation.

A micrometeorite hail, should it occur, is expected to be only transient in nature. Instantaneous response of the structure might be quite high. However, the long-time average structural response would be negligible.

4.3.1 Launch

Since little experimental data will be available for some time on the proposed Saturn booster system, it is necessary to make some estimate of the expected noise level external and adjacent to the manned space cabin.

It is known that the rms sound-pressure level for the Atlas missile booster system, providing 470,000 pounds of thrust, is about 141 db. This is at a point 75 feet above the rocket engine exhaust and would correspond to the location of the missile's nose cone. Allowing 3 db for atmospheric conditions and other uncertainties, this external value would be 144 db.

For a cluster of rockets, the total rms sound-pressure level (SPL) of the Saturn system is to the Atlas system SPL, approximately, as the square root of the ratio of their thrusts, therefore:

$$p^2_t = \left[\frac{(1.5 \times 10^6)}{.47 \times 10^6} \right] p^2 \text{ ATLAS}$$

$$p^2_t = 3.2\, p^2 \text{ ATLAS}$$

Therefore, at 75 feet:

$$SPL_{(SATURN)} = 20 \log_{10} \frac{(3.2p^2 \text{ ATLAS})}{(p^2 \text{ Ref})}$$

$$= SPL_{ATLAS} + 20 \log 3.2$$

$$= SPL_{ATLAS} + 10.1 \text{ in } \underline{db}$$

However, the space cabin is located 175 feet from the exhaust plane. Therefore, the level at 175 feet would be:

$$SPL_{175'} = SPL_{75'} - 20 \log_{10} \frac{(175')}{(75')}$$

Substituting from above:

$$SPL_{175'} = (SPL_{ATLAS} + 10.1)_{75'} - 7.4$$
$$= 144 \text{ X } 10.1) - 7.4 = 146.7 \text{ db}$$

or not much higher than that of the Atlas system. This agrees rather well with empirical predictions (Hilton, Mayes and Hubbard). These are based upon engines up to 200,000 pounds thrust extrapolated to those of higher propulsive power and contains data which applies to Project Mercury boosters. Although this estimate is subject to error, it is believed that it represents a fairly reliable figure.

Based upon transmission loss data through the thin-walled DISCOVERER vehicle and through other satellite and re-entry nose cones of GE construction, a safe estimate for noise attenuation through the APOLLO vehicle may be considered to be about 20 db. This is a conservative figure for over-all attenuation considering constant octave level band input. Using this number as a first approximation for the transmission loss of sound travelling into the interior of the space cabin, cabin noise levels should not reach above 127 db during the launch phase of the flight. The spectral distribution externally will probably be similar to that of the Mercury Space Capsule or that of the Big Joe Capsule (Figures I-4-17 and I-4-18). Most probably the spectrum will approximate that of Figure I-4-17, since larger boosters would furnish more power in the lower frequencies. Note that in these figures the over-all noise reduction is 32 db and 40 db respectively. The reduction at the lower frequencies is based upon the unique conical construction of the Mercury system, which furnishes a high degree of panel edge restraint and over-all panel stiffness. At the higher frequencies, the mass effect of this construction is effective in reducing noise transmission.

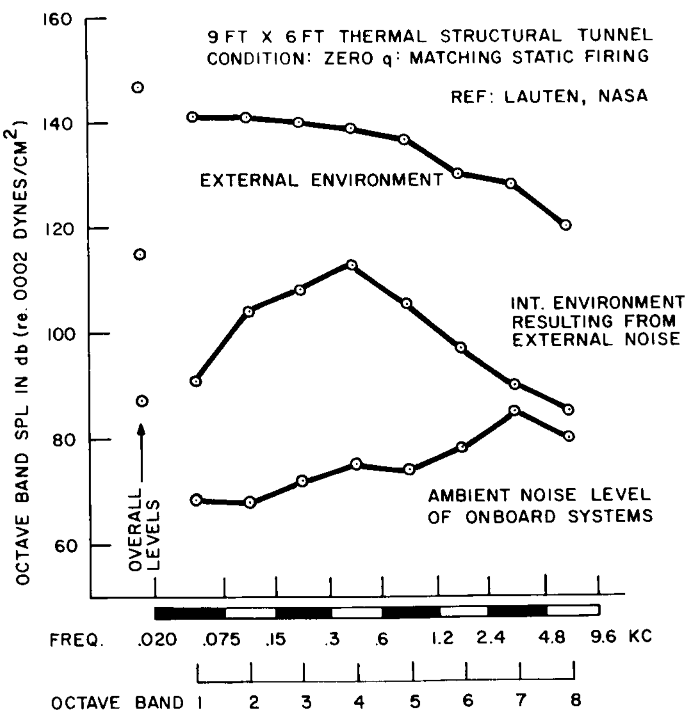


Figure I-4-17. Mercury space capsule measured external & internal noise environment

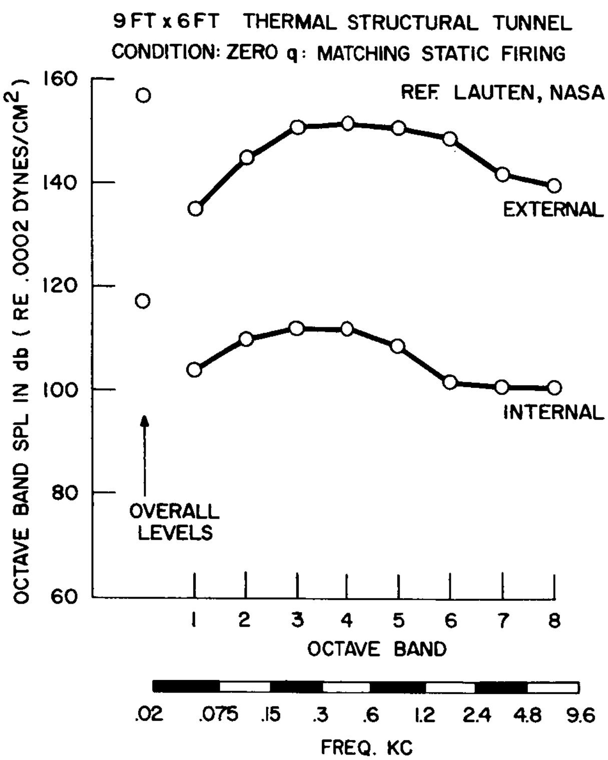


Figure I-4-18. Big Joe capsule measured external & internal noise environment

Considering the multiple panels of the APOLLO vehicle ranging in thickness between 0.032 and 0.040 inch of aluminum, the transmission loss, T.L., may be estimated from the empirical "mass law:"

T.L. = $23 + 14.5 \log_{10} m$ db where is the "mass" of the partition in lb/sq ft.

Therefore for a one sq ft panel of 0.040 aluminum:

$$m = \frac{.040''}{12} \times 1\ \text{ft}^2 \times .100 \times 1728 = .58\#/\text{ft}^2$$

$$\text{T.L.} = 23 - 3.5 = 19.5\ \text{db}$$

The behavior of a single wall is shown ideally in Figure I-4-19. A double wall behaves as in Figure I-4-20. At a particular frequency (fd) transmission loss drops because the air space, "d," between the walls acts as a spring whose stiffness varies inversely as the panel spacing. The panels behave as two masses, one on each end of the spring. The value of this loss depends upon damping in the cavity or in the panels. In this

[redacted]

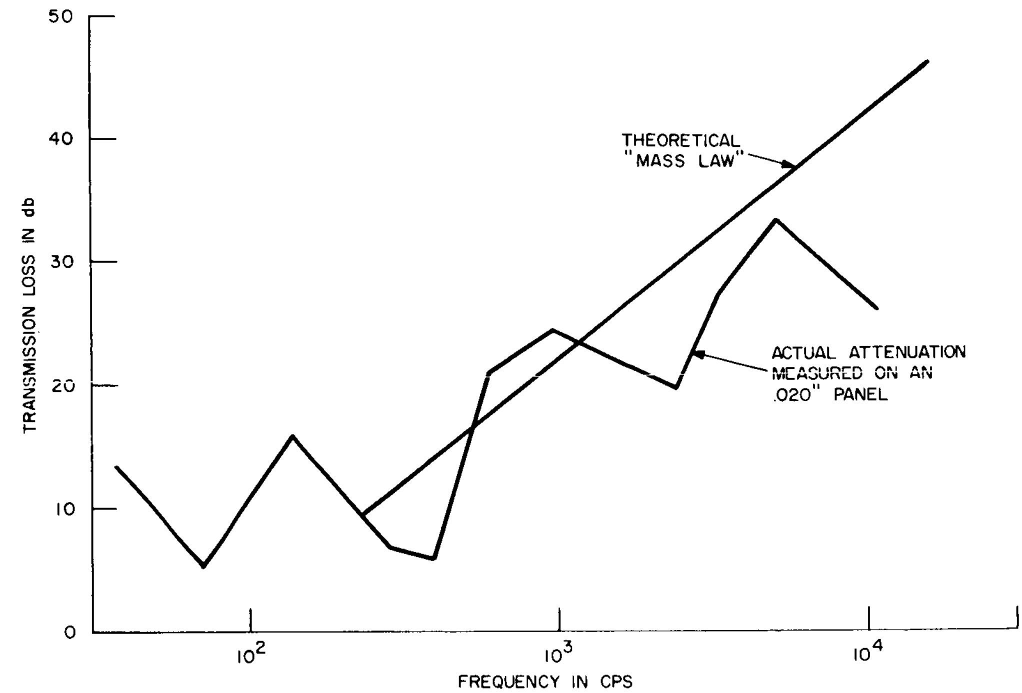


Figure I-4-19. Transmission loss vs. frequency single homogeneous panel

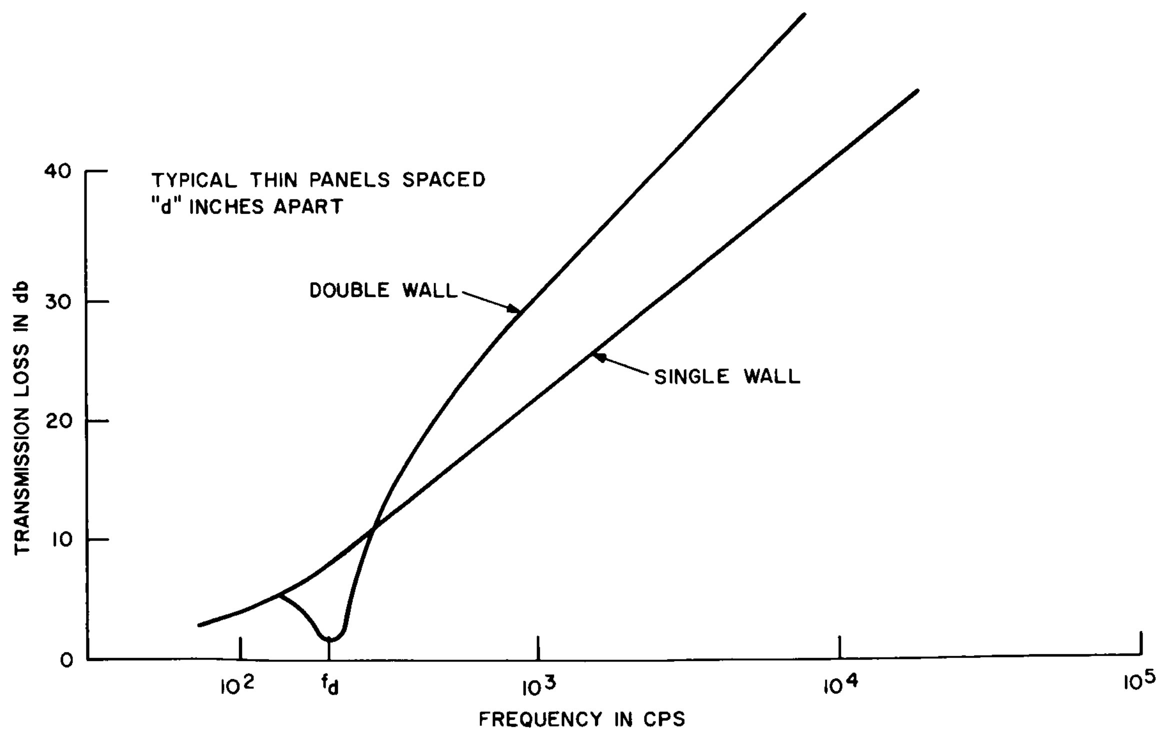


Figure I-4-20. Transmission loss vs. frequency double wall vs. single wall

[redacted]

configuration, since the panels are not parallel, nor of the same thickness, the behavior can be quite complex. For multiple construction, i.e., three or more panels in series, the complexities are even greater. For this reason, the engineering design should be based upon measured values of transmission loss. From the above discussion, it is evident that the transmission losses through the thin panels during launch are at least 20 db. In addition, noise reduction through the ablation shield, based upon the Atlas Mark II re-entry vehicle configuration (Heat Sink) and through the RVX and Atlas Mark III re-entry vehicle configurations are higher than 20 db. Measured values run from 23 to 28 db over-all (flat spectrum input).

From viewing many oscillogram records of missile flights, it is seen that the aerodynamic noise contribution to internal noise levels during launch lasts for a very short time, at most 5 seconds. The significant noise regimes during powered flight are those of Mach I and maximum dynamic pressure (or max. q). From the predicted dynamic pressure data during the boost phase of this flight, the boundary layer noise external to the manned cabin has been plotted as a function of dynamic pressure in Figure I-4-21. The estimated noise is based upon subsonic tests by Willmarth in which rms pressure fluctuations are equal to .006g. Then,

$$\text{SPL} = 20 \log_{10} \frac{(0.006\ g)}{P_{ref}}$$

$$\text{where } P_{ref} = 4.177 \times 10^{-7} \text{ psf}$$

Application of this equation to Mach number ranges of nearly 6.0 produces good results and so this equation will be retained.

Figure I-4-22 shows the variation of boundary layer noise with Mach number. As presented, this is an external estimate and it might therefore be assumed that the internal cabin values might be 20 db less than this curve. This is not the case. For, as the speed of the missile increases, the noise spectrum shifts toward the higher frequencies. The spectrum peaks at the high Mach numbers. However, as shown in Figures I-4-19 and I-4-20, there is more attenuation or noise reduction at the higher frequencies so that the internal noise is further reduced. Figure I-4-23 shows the internal measured spectra of the Big Joe vehicle during powered flight at two different Mach

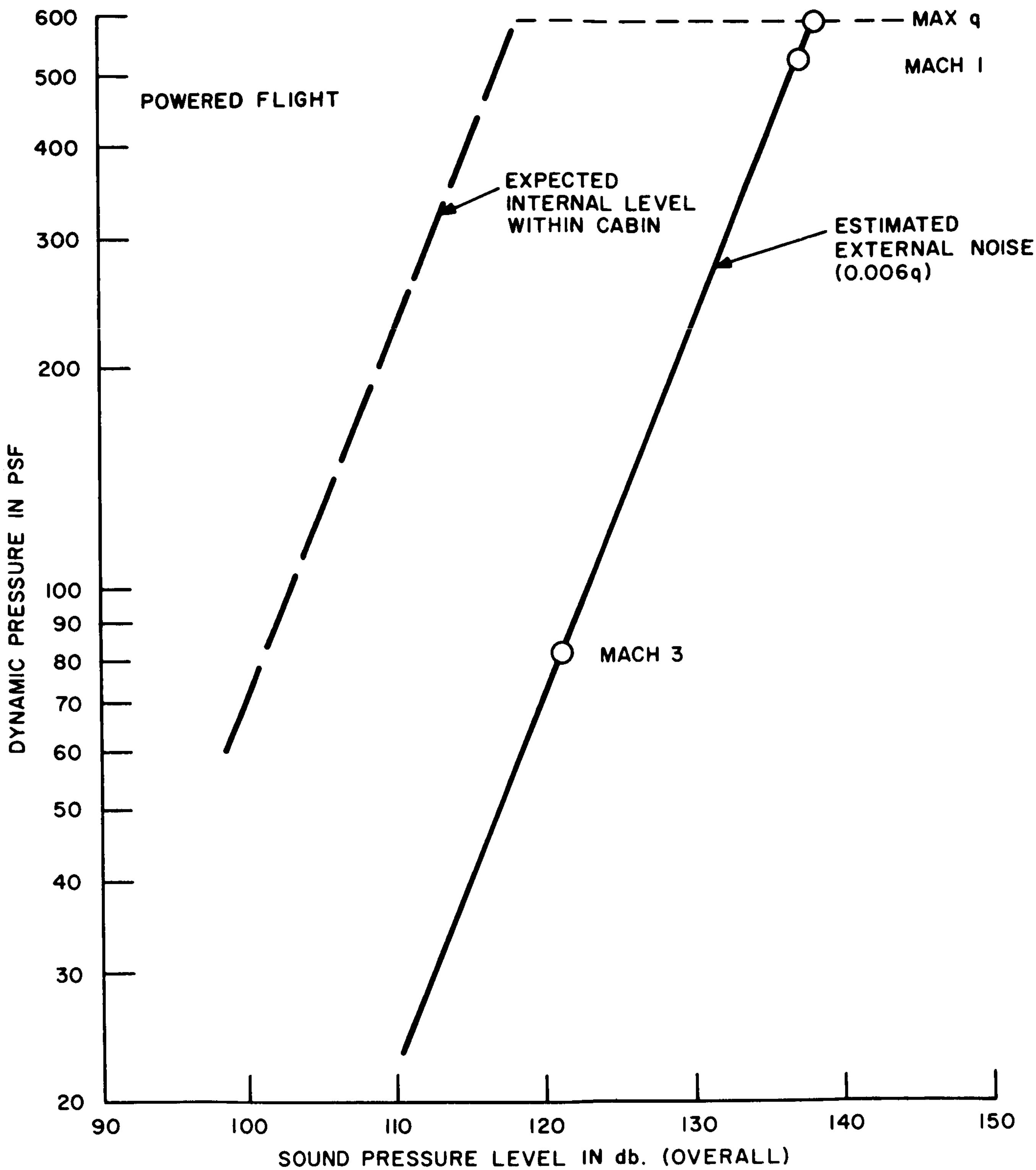


Figure I-4-21. Estimated external noise aerodynamic boundary layer turbulence

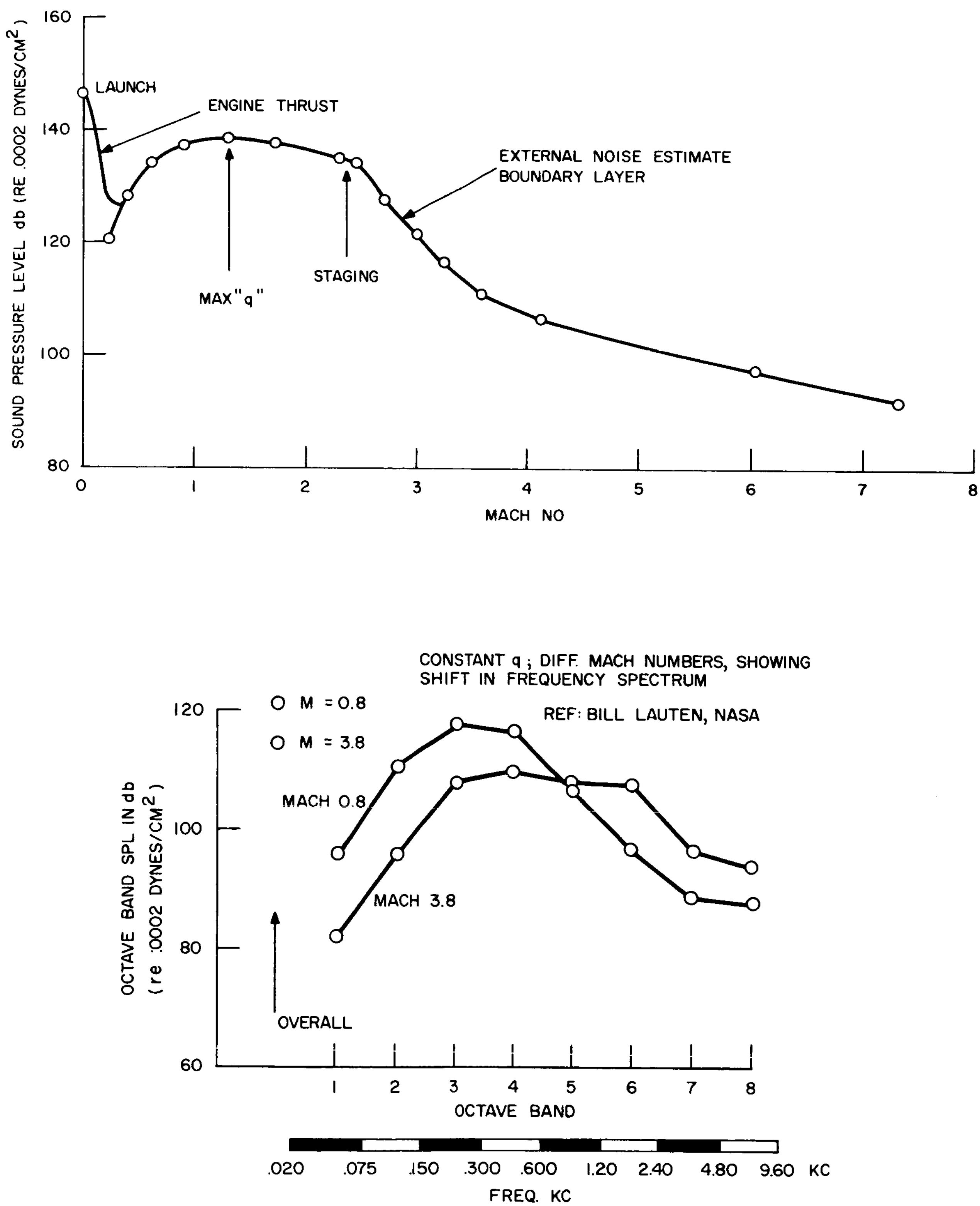


Figure I-4-23. Big Joe capsule measured internal noise data powered flight values having the same dynamic pressure. This produces, theoretically, the same external over-all noise level, but internally, a reduction of 6 db, or a reduction in intensity equal to one quarter of the sub-sonic value is achieved. It should be noted that these are internal spectra; the external spectra would be expected also to shift in this manner and probably by a greater amount.

4.3.2 Re-entry

The severest re-entry trajectory was chosen for the D-2 configuration in order to arrive at the combination of velocity, pressure, density, and other parameters which would produce the maximum sound pressure. This information was fed into a digital computer which corrected for, among other factors, Mach number compressibility effects and local flaw conditions. As part of the necessary input information for the turbulent wake noise predictions, a plot of the base-pressure to free-stream pressure ratio versus Mach number was calculated. Furnished with the necessary input data and the appropriate program, the computer printed the necessary data points along the re-entry trajectory.

Figure I-4-24 shows the external noise prediction for boundary layer and wake excitation in decibels as a function of the re-entry altitude. Cabin levels are again assumed to be 20 db below these figures.

4.3.3 Summary

Based upon the above analysis it is possible to draw certain conclusions about the noise level problem with respect to the crew of the APOLLO vehicle. The first conclusion is that since the maximum internal noise level occurs during launch and since this is not expected to exceed 127 db, there appears to be no requirement for the inclusion of special sound adsorbing materials in the cabin structure. This is particularly true when considering that only 20 db attenuation has been assumed even though evidence exists which might logically permit values as high as 28 db or higher to be used.

The second conclusion is that when an additional 10 to 12 db of attenuation is added as a result of the helmet characteristics, the net 117 db level which reaches the inner ear is well within physiological limits as described by O'Connell. See Table I-4-III and Figure I-4-25. Furthermore, this db level will place no unusual constraints upon

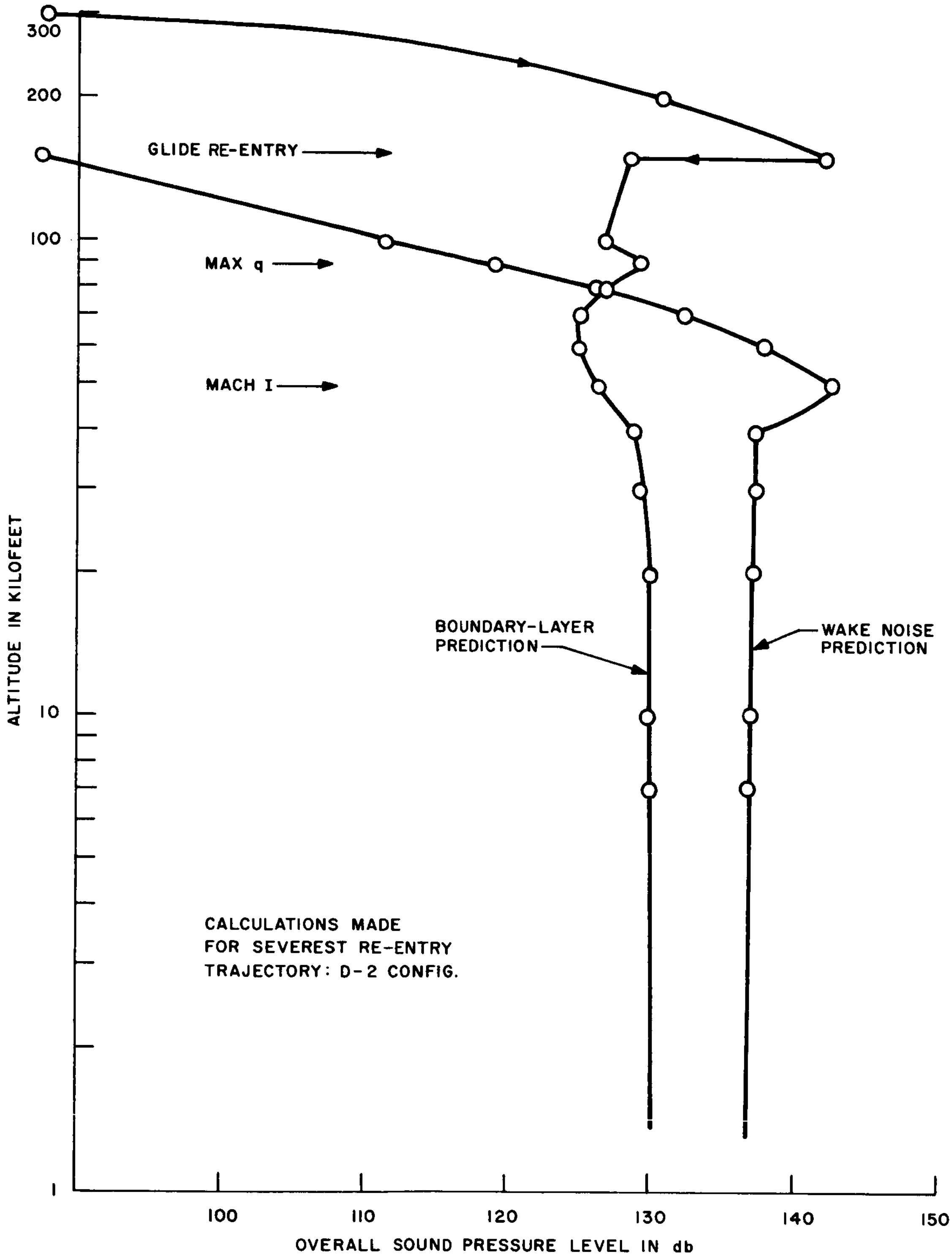


Figure I-4-24. APOLLO re-entry vehicle external noise prediction for boundary layer & wake excitation

TABLE I-4-III. HEARING DAMAGE RISK CRITERIA

Duration of Daily Exposure	No Protection	Ear Plugs Only	Ear Muffs and Plugs
8 Hours	100 db	112 db	120 db
1 Hour	108	120	128
5 Minutes	120	132	140
30 Seconds	130	142	150

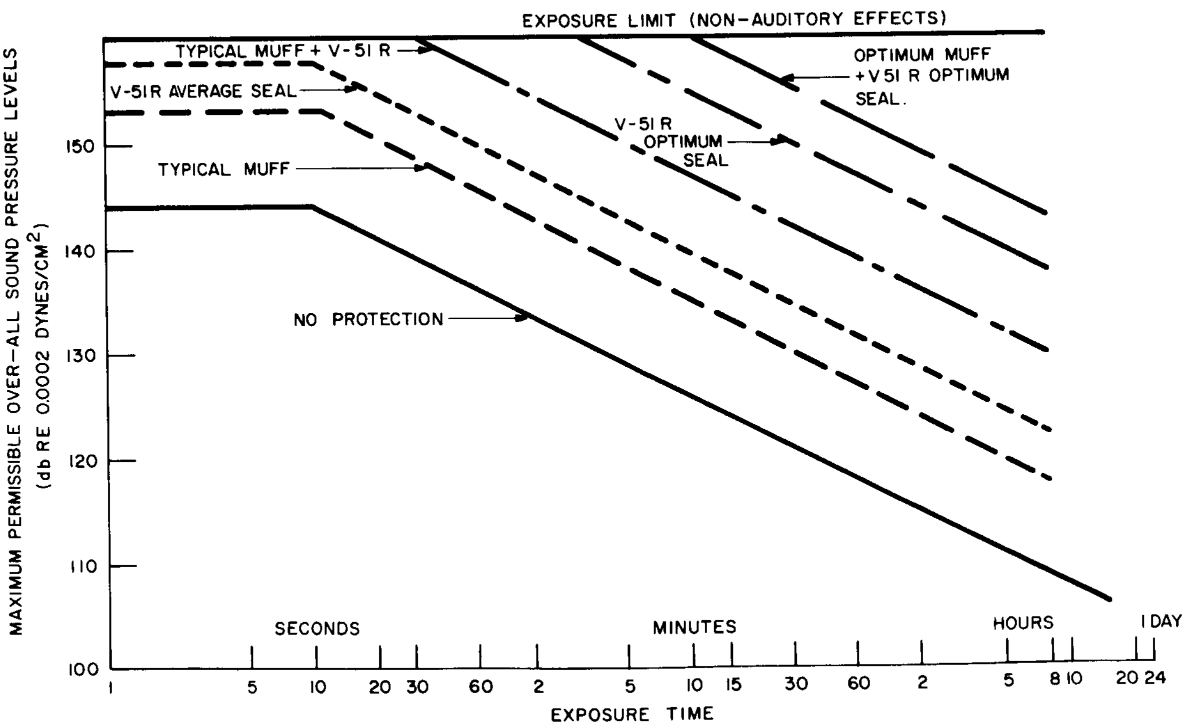


Figure I-4-25. Physiological limits of sound level the design or use of the communications system. The 10 db speech to noise ratio desirable can be easily achieved with present state-of-the-art equipment in this acoustic environment.

4.4 VIBRATION TOLERANCE

The occurrence of transient mechanical vibrations expected during the launch and reentry phases of APOLLO flights requires serious consideration as a potential safety hazard to the crew. However, no general prediction scheme exists for the evaluation of structure-borne vibration because the frequency and levels depend markedly on the specific details of engine and stage structure as well as the aerodynamics of the flight phase.

Although over-all vibration spectra for particular missile systems may not be known for large vehicles, low-frequency resonances do occur (Figure I-4-26). Although few data are available on launch and in-flight vibration levels that extend to such low frequencies, von Gierke concludes that the vibrations from booster engines constitute no

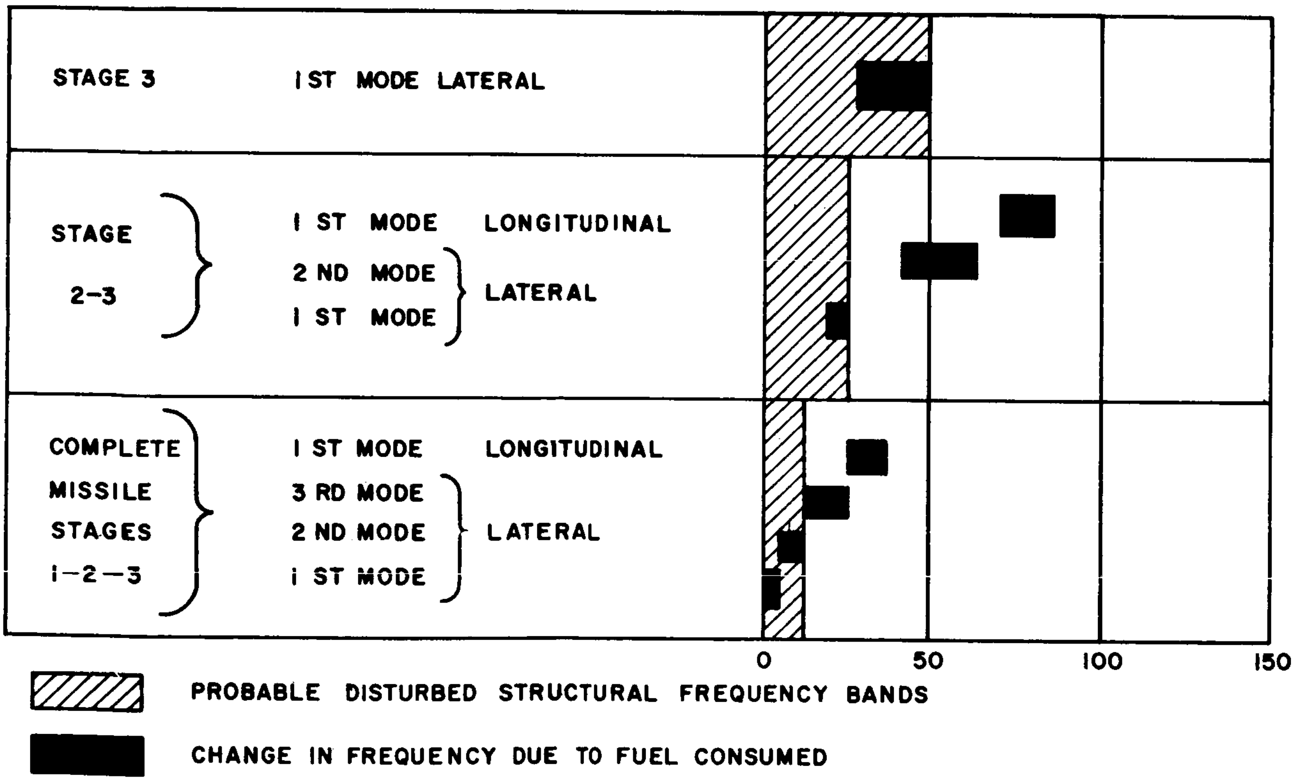


Figure I-4-26. Missile vibration spectrum serious hazard to man, principally because of their short duration. Greatest severity is expected when the missile shudders on the pad at the moment of launch. The transverse low-frequency caused by corrections of the guidance system probably will generate only 1 g or less and will be amenable to engineering control.

The resonances of importance in the body structure of man are in the thorax-abdomen system. For a constant acceleration input abdominal acceleration is plotted in Figure I-4-27 for longitudinal vibration on a shake table. Peak resonance occurs at approximately 6 cps, although other resonances occur at 3 and 10 cps. Vibration tolerance depends strongly on the direction and the nature of the body support and restraint harness employed. Little or no data are available for complex vibrations in multiple axes. Ziegenruecker and Magid have accumulated tolerance data for humans from the literature, as shown in Figure I-4-28, for continuous sinusoidal vibration parallel to the long axis.

Since vibration data are not presently available for the Saturn system, particular attention must be accorded all future Saturn and upper stage flight and static firings.

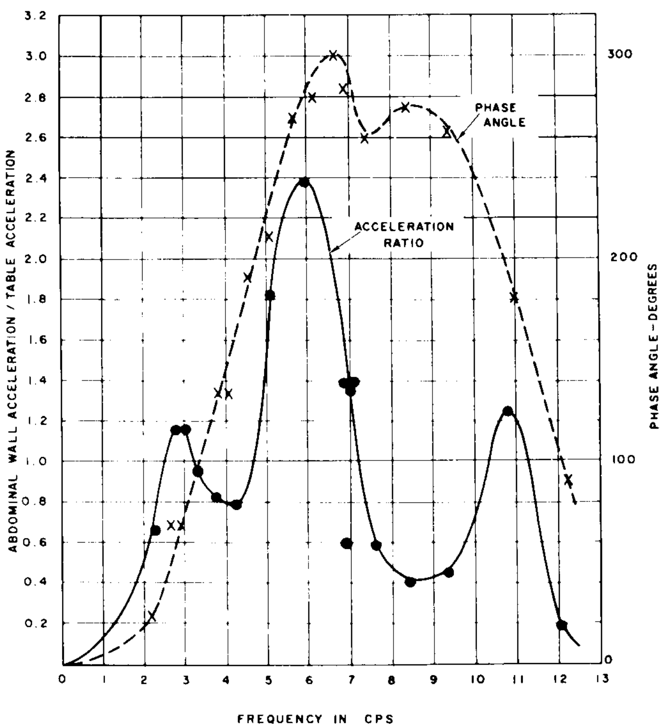


Figure I-4-27. Abdominal wall resonances of supine human vibrated along long axis.

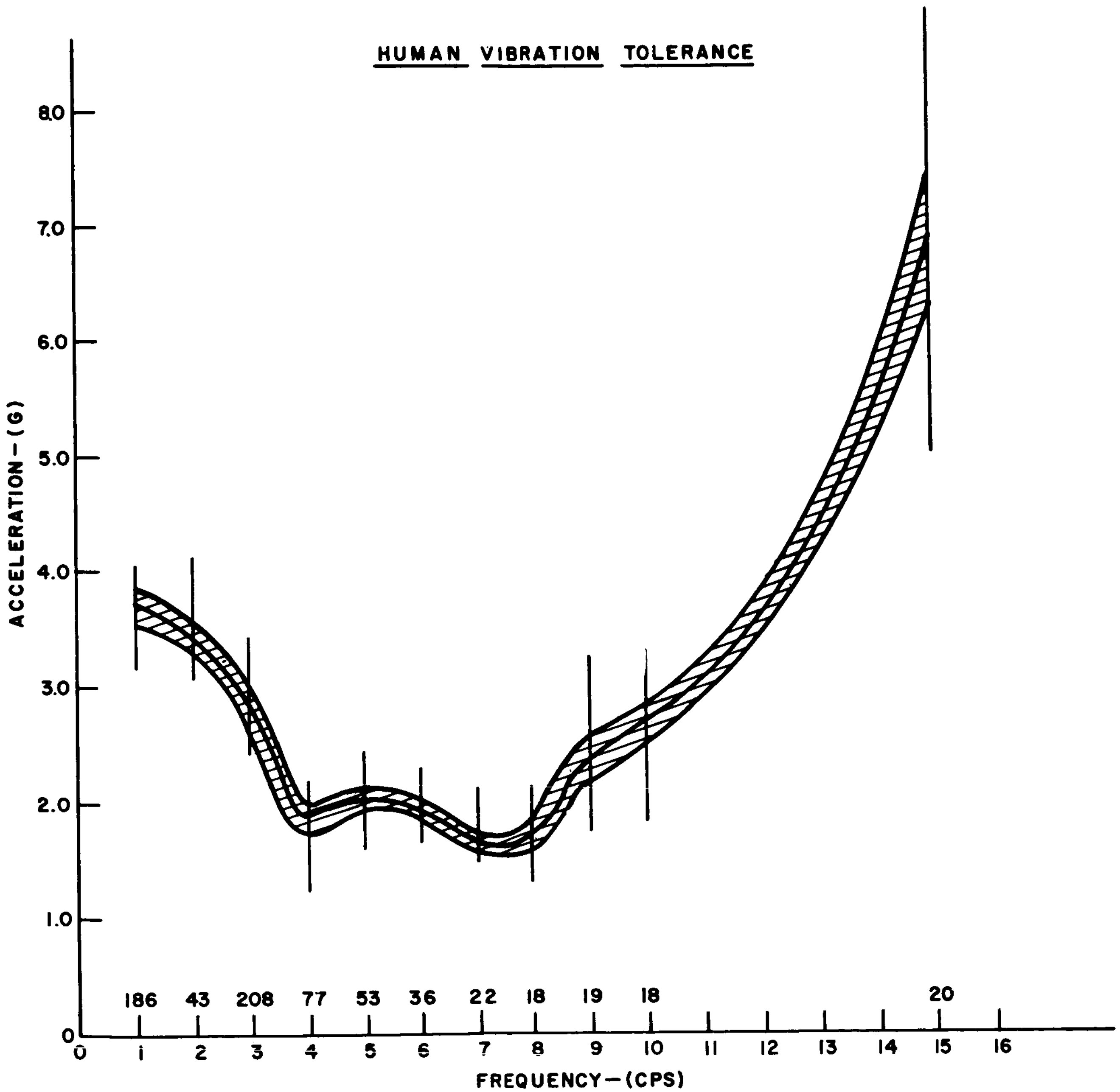


Figure I-4-28. Average peak acceleration at various frequencies at which subjects refuse to tolerate additional exposure to vertical vibration. The figures above the abscissa indicate the exposure time in seconds at the corresponding frequency. The shaded area has a width of one standard deviation on either side of the mean (10 subjects).

It is hoped that instruments will be mounted to these units on a noninterference basis such that the data necessary to completely assess the vibration spectrum can be obtained. If this is accomplished early enough, it will permit integrated damping to be incorporated into the vehicle design should this damping be required.

Once the spectrum and profile are established, a program can be conducted to measure both tolerance and performance of the crew within this environment. The new vertical accelerator in the Biacoustics Laboratory at the Wright Air Development Division, Dayton, Ohio, is suggested as a possible location for conducting these experiments.

5.0 Food, Water, Waste and Personal Equipment

5.1 METABOLIC CONSIDERATIONS

The 80th-percentile man may be taken for the purpose of determining man's metabolic inputs and outputs. It appears improbable that the average size of the three men who will eventually man Project APOLLO will be greater than the 80th percentile. Such a hypothetical man has a surface area of 2.02 square meters. Since heat production, oxygen consumption, carbon dioxide production, and caloric intake are dependently related to the level of metabolism, an estimate of this level and its probable ranges is required. The unit of metabolism is the MET, where 1 MET = 50 $KCal/m^2/hr$. For a work-rest-sleep cycle in which eight hours is given to each phase and, at any given time (exclusive of exit, re-entry, and emergencies), one man is sleeping, one man is resting, and one man is working, it may be assumed that the average is approximately one MET. The rationale for such an assumption is that during rest (8 hours), one MET may be assumed as the level of activity, during sleep (8 hours) 0.7 MET may be assumed, and during work 1.2 MET may be assumed. 8 hrs. at 0.1 MET should be sufficient to account for exercise. Of these three levels only the work MET level may be open to question. However, it appears probable that movements in a zero gravity field will require a lower MET level than in a one-g field. Furthermore, movements may be somewhat restricted in space cabins such as those under consideration for Project APOLLO. Experimental evidence supporting this concept has been obtained at the USAF School of Aviation Medicine in their one- and two-man simulators. The metabolic level range varied from 1800 to 2400K-Cal/man/day. The lower MET level may be taken as 0.7 MET (sleep) and the upper level may, for short periods, be as high as 6 MET. The environmental control system should be capable of maintaining steady state conditions should such extremes occur.

Thus, the four dependent variables may be calculated on the basis of one MET average. (See Table I-5-I - Metabolic Requirements.) Ranges may be obtained by multiplying these average values by 0.7 and 6.0.

TABLE I-5-I - METABOLIC REQUIREMENTS

Heat Production	2.02 x 50 = 101 K Cal/man/hr = 404 BTU/man/hour.
Oxygen Consumption	Assume an R. Q. of 0.82 and a caloric equivalent of 1 liter of oxygen = 4.8 K Cals. Then, $\frac{101}{4.8}$ = 21 liters 0_2/man/hour 1.6 lb 0_2/man/day.
Carbon Dioxide Production	21 x 0.82 = 17.2 liters CO_2/man/hour 1.8 lb/man/day.
Caloric Intake	24 x 101 = 2424 K Cal/man/day. The composition should be approximately 52 percent carbonhydrate, 33 percent fat, and 15 percent protein and contain all essential vitamins and minerals. Dry food weight 1.0 lb/man/day.
Water Balance	a. Output = 5.51 lb/man/day Urine: 1500 ml/man/day = 3.31 lb/man/day. Feces: 150 ml/man/day = 0.33 lb/man/day. Skin and Lungs: 850 ml/man/day = 1.87 lb/man/day. b. Input b. Input = 4.85 lb/man/day. Drinking water and water in food: 2200 ml/man/day = 4.85 lb/man/day. Water of food oxidation: 300 ml/man/day = 0.66 lb/man/day.
Solids = 0.2 lb/man/day	Urine: 55 - 70 gm/man/day Sweat: 3-8 gm/liter Feces: 30-80 gm/man/day.
Input =	Output = 7.5 lb/man/day

5.2 FOOD

When choosing a nutritionally adequate diet for any mission, the factors of weight, volume and acceptability are the most critical parameters to be considered. A meal that is of low weight and volume, but highly palatable is required. A highly acceptable diet is considered to be an extremely important factor in crew morale and performance.

In order to minimize weight and volume, dehydrated foods are utilized wherever possible. To increase the acceptability of the diet, canned and frozen foods also may be included. A nutritional level of approximately 2500 kilocalories per day has been suggested, 15 percent of which is obtained from protein, 33 percent from fat, and 52 percent from carbohydrate.

Menus have been designed for three levels of acceptability. They are designated maximum, intermediate and minimum.

The maximum acceptability diet is naturally the most desirable with a high nutritional level and a wide variety of foods including snacks and many between meal beverages. Appetizers are provided in canned and dehydrated forms while the main dishes and vegetables are frozen or canned. Desserts, bread, cake and various items such as candy, nuts and sandwiches are offered in canned or frozen forms. Beverages and soups are in the dehydrated form and are reconstituted by the addition of hot water. All the foods are precooked and can be heated prior to consumption. Thus, an oven is required. The canned, dehydrated and frozen foods are stored in separate containers; one container per portion. A refrigerator is required for the frozen foods and beverages, juices and fruit can be made more acceptable by the inclusion of a chill compartment in this refrigerator.

The suggested diet provides food at the rate of 3.20 pounds per man day and water at 6.0 pounds per man day, 1.65 pounds of which is contained in the food.

Menus have been developed (ARDC Technical Report No. 60-8, July 1960) and are listed in Table I-5-II, a representative one day menu of a maximum acceptability diet.

TABLE I-5-II - MAXIMUM ACCEPTABILITY DIET

Breakfast	Lunch	Supper
Orange Juice	Tomato Juice	Vegetable Soup
Oatmeal	Roast Beef and Mushroom Gravy	Tuna and Rice
Scrambled Eggs	Buttered Peas	Buttered String Beans
Bacon	Baked Potatoes	Baked Apple
Bread-Butter	Bread-Butter	Bread-Butter
Milk	Butterscotch Pudding	Chocolate Chip Cookies
Coffee	Coffee or Tea	Cocoa or Milk

The intermediate acceptability diet differs from the maximum in that no frozen foods are provided and some of the supplementary beverages have been deleted. This narrows the variety of foods somewhat and reduces the water consumption. However, a high nutritional level is still available and sufficient water for the metabolic processes is provided. Canned and dehydrated appetizers are provided as in the maximum acceptable diet. The main dishes, vegetables and desserts are in the canned and dehydrated forms and thus may not be as appetizing as the frozen form. (Dehydration by freeze-drying indicates excellent flavor and texture retention.) The foods are precooked and only require heating prior to consumption. The dehydrated foods require the addition of hot water. Since frozen foods have been eliminated, there is no need for a refrigerator, but a means of reducing the temperature of beverages such as milk and juice can be provided by utilizing a cold region of the air conditioning system. The previous menu is also an example of the intermediate acceptability diet (Table I-5-II).

The food is provided at the rate of 3.08 pounds per man day and water at 5.0 pounds per man day, 1.64 pounds of which is contained in the food. The weight of the food required for this diet is slightly less than that of the maximum diet, the water consumed is reduced by 1.0 pound per man day, and the weight of the refrigerator and associated equipment has been eliminated. The only penalty which appears necessary occurs through the reduction in the variety of food which can be supplied. This is not to imply that no variety is possible and considering the 14 day maximum mission duration for APOLLO this compromise can not be assumed to be critical.

The minimum acceptability diet is one that involves the use of practically all pre-cooked dehydrated foods supplemented with sandwiches, food bars and bite-sized solid foods. The addition of hot water is the only requirement necessary for reconstitution. Instead of the wide variety of foods as shown in the previous diets, only a relatively narrow range of foods is available. The problem of food monotony, a lower level of food acceptability and probable resulting decrease in food consumption are considerations in this case.

For such a diet, the only components, in addition to the food and water, which are necessary are storage compartments and a water heater. The weight, volume and power requirements of this diet thus are reduced to a minimum. The food is provided at the rate of 2.15 pounds per day and water at 4.85 pounds with 1.0 pound contained in the food.

Although the minimum acceptability diet represents a decrease in both food and water consumption over the intermediate acceptability diet, it does satisfy minimum nutritional requirements.

Several curves have been constructed to be used in estimating the weight and volume of nutrients needed for missions of up to 70 man day duration. Figure I-5-2 is the volume equivalent of Figure I-5-1. It must be pointed out that both of these curves include a weight of water plus container which assumes that no water recovery system is used aboard the vehicle. This problem is discussed in the section on water supply.

The diet recommended for use in the APOLLO system is that described as intermediate. For a two week mission there appears to be little compromise between this and the maximum acceptability menu. The refrigerator and the between-meal snacks and extra beverage consumption called for in the maximum diet will weigh an additional 85 pounds. This is certainly excessive for the benefits which may be gained. Appendix HF-J discusses the design and weight penalty of a refrigerator suitable for use in this vehicle should refrigeration be desired.

On the other hand, while the minimum diet and associated equipment will weigh 35 pounds less than the intermediate the low order of acceptability and subsequent food rejection may have deleterious effect upon morale and performance of the crew.

Since the crew must be used to maximum advantage in this vehicle a high level of performance should not be sacrificed by adding the additional stress of food monotony.

Therefore, the intermediate diet has been selected as the best compromise between weight, volume and acceptability for this mission.

5.3 WATER

Considering the diet chosen, the ranges of temperature and humidity expected and the anticipated activity level, a water balance has been calculated which can be used as the basis of design of the water supply system for the APOLLO vehicle.

This daily water turnover is somewhat different than that shown in Table I-5-I, since the diet to be used requires a slightly increased water consumption than the minimums expressed above. Our purpose in repeating this is to illustrate how little additional water is required even though a highly acceptable diet is chosen.

Input	
Consumption (food and drink)	5.00 lb/man
Water of food oxidation	.66
Total input	5.66 lb/man
Output	
Urine	3.43 lb/man
Skin and lungs	1.90
Feces	.33
Total output	5.66 lb/man

Since, from above, the water content of the food consumed is 1.64 lb/man/day, the supplemental water required is 5.00-1.64 = 3.36 lb/man/day. Therefore, for the three man, fourteen day mission a total of 141 pounds of water is required for support of the crew. This large quantity has obviously led to an investigation and trade-off of water recovery systems which might be feasible for use in the APOLLO vehicle.

The three types of water output, that in the urine, from the skin and lungs and that contained in the feces were investigated separately since their level of impurity and hence the method of treatment would not necessarily be identical. Feces were immediately eliminated from consideration since, for this mission profile, the recoverable water total is only 14 pounds and since the problems of handling, extraction and purification do not appear practical within a reasonable weight penalty.

Water from the skin and lungs was considered next. This water fraction may be collected as condensate on the cabin thermal control heat exchanger and can readily be transported to a storage tank. Experiments conducted by the crew of the submarine Skipjack who for three days drank the condensate collected on the air conditioning heat exchanger indicate that this water is potable and requires no further treatment for human consumption. Furthermore, by careful selection of heat exchanger materials, it is expected that the metallic taste reported by the investigators will be largely eliminated. A small charcoal filter placed between the collector and storage tank is expected to remove any additional objectionable odor which might be carried over. Since the weight of the collecting device and filter is estimated at only four pounds and since 3 x 1.9 = 5.7 pounds of water can be recovered each day it is obvious that the system pays for itself very quickly. Thus, only 1.46 lb/man/day (3.36 - 1.90) of additional supplemental water is required.

Since about 3 pounds of urine is available per man/day an investigation and trade off of urine recovery methods was made. Several systems of this nature which have demonstrated laboratory feasibility are available for consideration. The first system analyzed is a vacuum distillation method wherein the urine vaporized at reduced pressure and then condensed, collected and passed through a charcoal filter. While encouraging, the results of the work done with this system indicate that a relatively large percentage of ammonia passes over with the water. In addition, the condensate is sometimes turbid and of a slightly yellow color. There also exists the possibility of bacterial transfer. For although urine is generally considered sterile it has the capability of supporting bacterial growth. These objections obtain considerable significance when considering that the crew urinary output will be recycled and reconsumed as many as fourteen times during a mission.

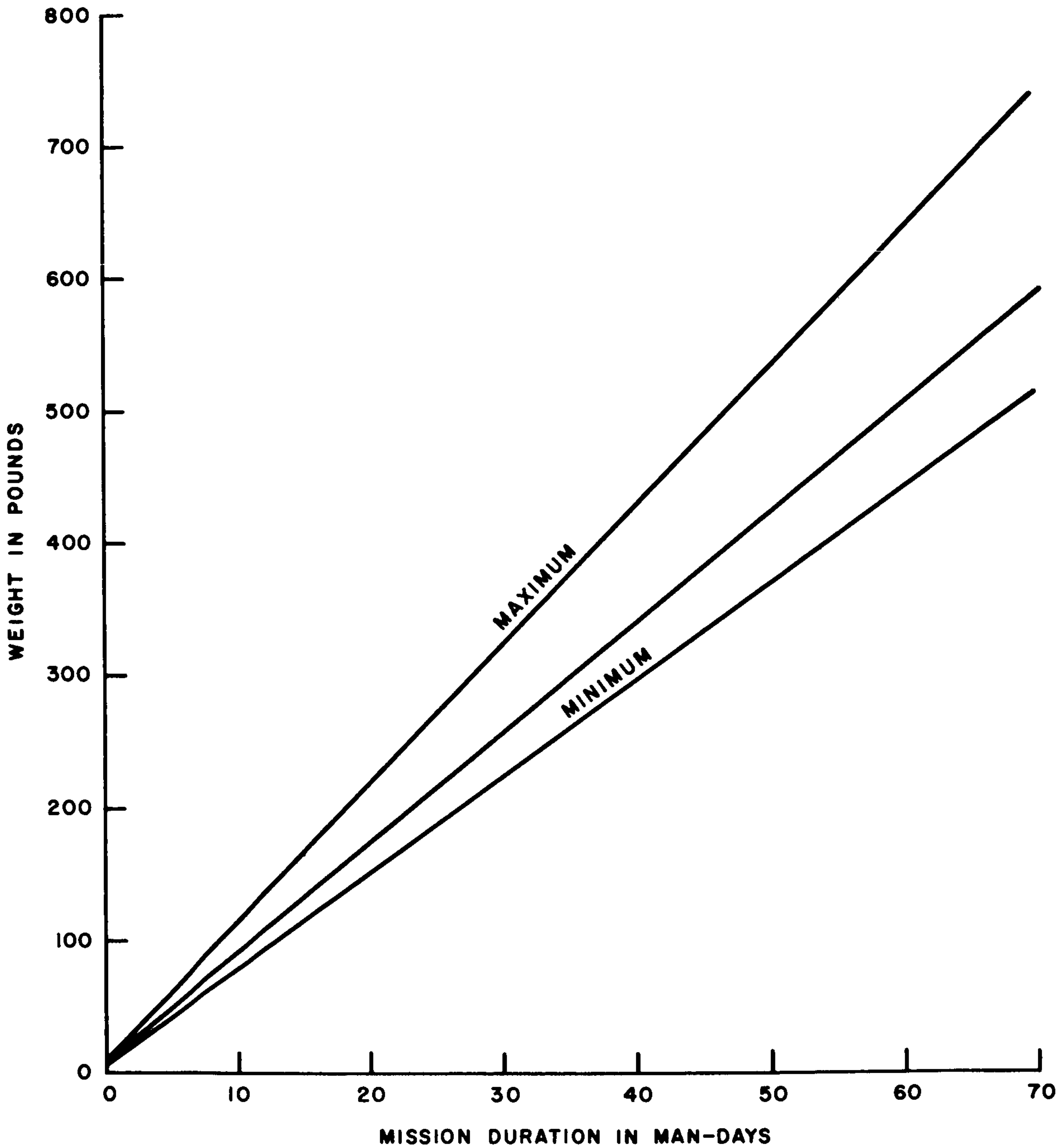


Figure I-5-1. Nutrient weight vs mission duration
For maximum, intermediate and minimum acceptability diets

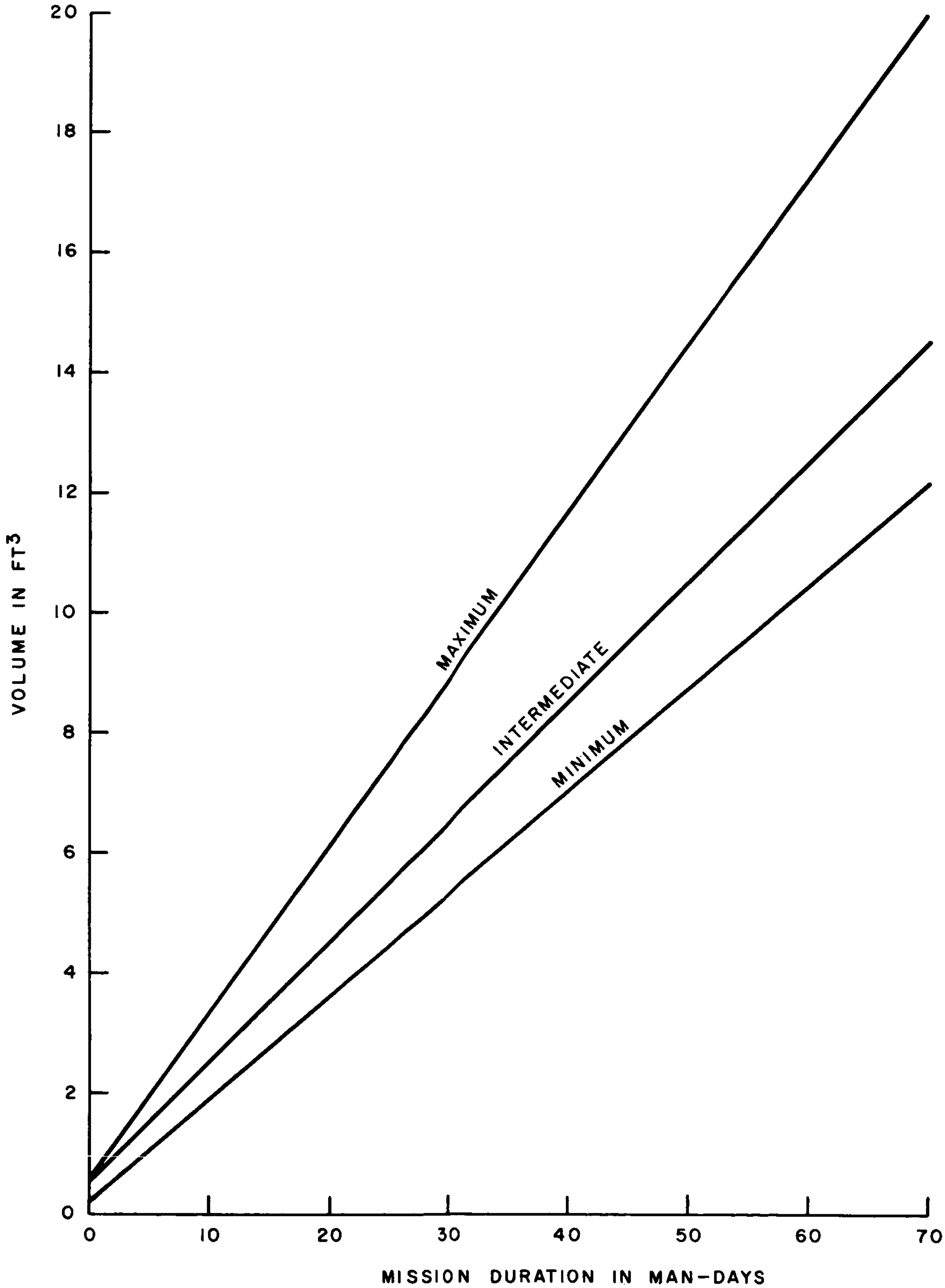


Figure I-5-2. Nutrient volume vs mission duration
For maximum, intermediate and minimum acceptability diets

As a result of these considerations it is felt that simple vacuum distillation is unsatisfactory as a primary method of water recovery. However, by the addition of a pyrolysis cycle to the vacuum distillation process, completely potable water can be produced. (See Appendix HF-L.) Pyrolysis breaks down the ammonia and destroys any bacteria which might be present. A laboratory model of this apparatus has been constructed and is in operation at MSVD, General Electric Company. The only apparent disadvantage of this system is the power required to perform the pyrolysis operation.

A third method of water regeneration which has been investigated by the General Electric Company employs the use of regenerable ion exchange fuel cells, a General Electric Company development, to first electrolyze and then recombine the water content of the urine. (See Appendix M.) With the application of power the fuel cell (electrolysis cell) will electrolyze the water and separate the oxygen and hydrogen. The recombination fuel cell then recombines the O_2 and H_2 to produce water and recovers approximately 50 percent of the power initially required for separation. Tests conducted at MSVD, General Electric Company have indicated the feasibility of this concept.

In order to evaluate the use of either of these methods it is necessary to trade-off the weight of equipment and power requirements against the weight of carrying the water as an open supply. This comparison has been made and is illustrated in Figure I-5-3. This curve shows for the APOLLO mission duration of 14 days and a power penalty of 125 pounds per kilowatt in sun-only operation and 200 pounds per kilowatt in over-all operation that urine recovery is not practical. Shown in dashed lines, however, is the vacuum distillation method of urine recovery which can be used as a secondary source of water for survival considerations. The system of recovering water vapor from cabin air is also shown for reference purposes.

Provision of water for the fourteen day mission profile is not sufficient compliance with the total supply requirements. In addition, water is required as a portion of the survival equipment necessary to sustain the life of the crew for three days after an unprogrammed landing in desolate country. The most severe criterion for this water supply is generated by the possibility of this landing occurring in a desert area when the requirements for water can reach 6 to 8 pounds per man per day, assuming no travel or travel at night only. This problem is particularly real not only because of

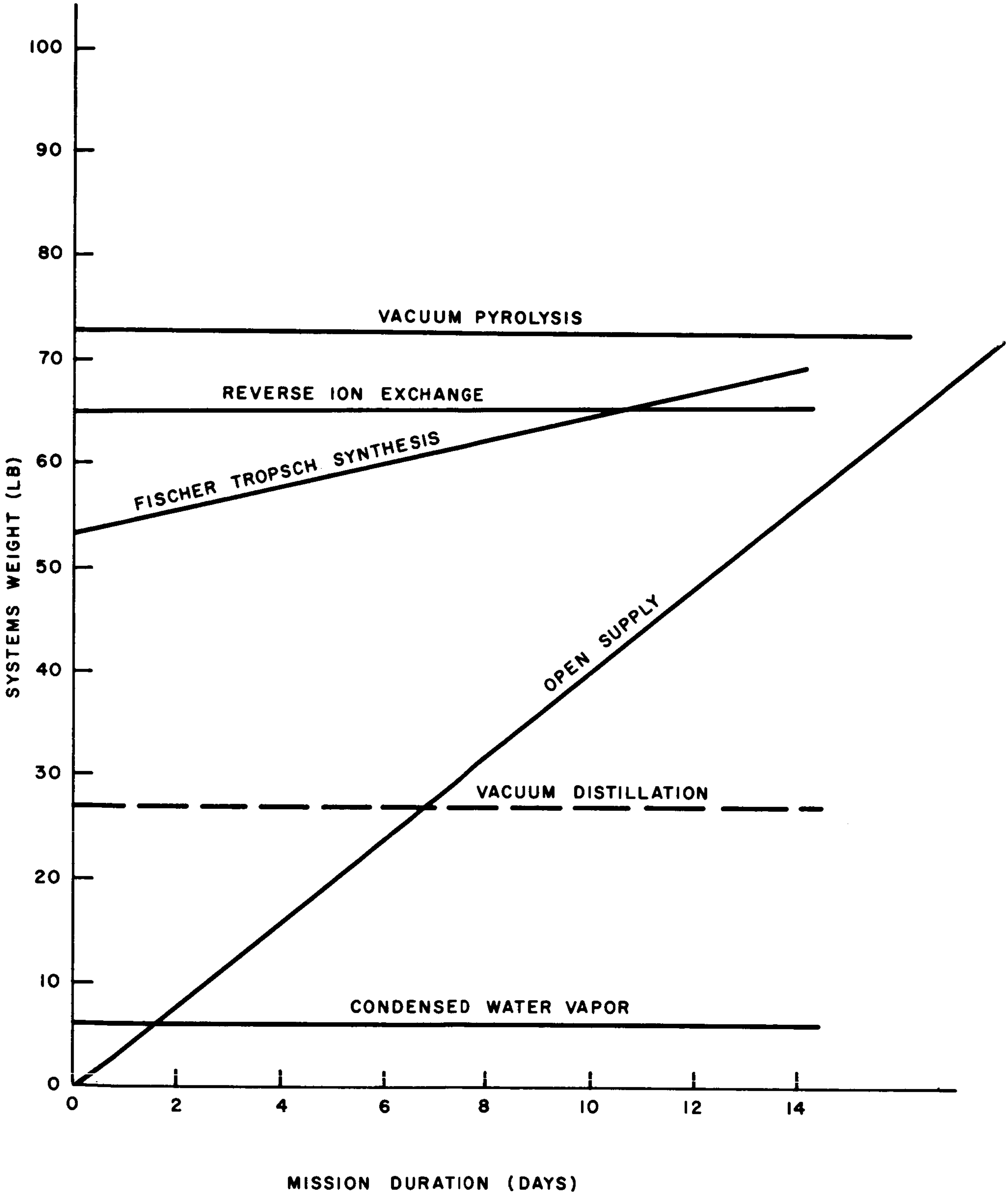


Figure I-5-3. Weight to recover 4.8 pounds of urine per day vs mission duration the large weight involved (63 pounds of water plus container) but also because of the possible choice of the Woomera Range in Australia as a secondary landing sight.

Analysis of this problem, however, has resulted in an acceptable solution at minimum weight. The only method of primary water recovery suitable to this mission is that returned through the humidity control system. Therefore, an additional 1.46 lb/ man day x 42 man days or 62 pounds of water must be carried as open supply. In the case of an abort resulting from a down-range launch malfunction, earth impact occurs prior to the time when any water could have been consumed. As a result, enough stored water to satisfy the desert survival requirement will be available. In the event, however, of either an emergency return during a later phase of the mission, or of an unprogrammed landing as a result of an error during re-entry, water equal to that consumed from storage must be provided. This is an emergency mode and, since this water will be consumed only once, the simple vacuum distillation method of urine recovery is satisfactory.

Therefore, if a vacuum distillation system were included in the vehicle and designed to recover 1.46 lb/man/day x 3 men or 4.38 lb/day of water, then regardless of time of impact approximately 62 pounds of survival water would always be available. Water recovered from urine would be kept in a separate tank in the mission module and transferred into the primary water container prior to re-entry. In this manner a minimum of 60 pounds of consumable water regardless of time of impact is assured. The weight of the system required to provide this capability is estimated at 27 pounds (Figure I-5-3).

The total water supply system, therefore, includes an open supply supplemented by humidity recovery equipment and a urine recovery system for make-up to be used as survival water. The required 60 pounds of post-impact water thus are supplied at an equipment weight of only 27 pounds.

5.4 GALLEY DESIGN

The galley will be installed in the mission module and will provide all the food and water storage equipment, heaters, service devices and sanitation aids necessary for the preparation of meals and snacks. The galley will be integrated with the mission module equipment layout and will be designed such that minimum effort is required to perform the functions necessary for preparation and cleanup.

The food will be stored by portion in either 6 ounce squeeze bottles or 3 ounce "toothpaste tube" shaped containers. The tubes will contain such foods as meats, fruits, dehydrated potatoes, etc., while the squeeze bottles will be used primarily for beverages such as coffee, milk or juices. These containers will be stored on appropriately designed racks which will support the food during launch accelerations. The water will be carried in a container which is pressurized by a flexible bladder. This container will be located in the command module during launch and re-entry but will be transferred to the galley during cislunar and lunar orbit flight. The water recovered from the humidity control system will also be routed to this container for storage. The on-board nitrogen supply will be used to pressurize this bladder periodically to make-up for the volume of water used during the duration of the mission. Two needle-type valves, one for hot, the other for cold water, will be provided which will be inserted into the food containers for introduction of water. The distribution system will also be gated such that the proper quantity of water is provided for each bottle or tube. A flash water heater is provided between the two gates of the hot water circuit such that reconstitution of dehydrated foods can be achieved upon demand.

Sealable plastic bags will be provided within which the used food containers will be placed such that odor permeating the cabin will be minimized. An "out-of-sight" storage area will also be provided for these waste containers. Trays and zero 'g' feeding aids will be made available which will attempt to minimize the frustrations of consuming food in this unusual environment.

The galley will also be the personnel hygiene center in the vehicle. No sanitary water is provided in the vehicle system since the ultra-clean conditions and the relatively short mission duration do not make total body cleansing mandatory. Dry wash pads, similar to those used on aircraft, will be provided for hand and face washing and freshening up.

A paste dentifrice and brush will be provided for oral hygiene and a battery-powered electric razor for shaving. While neither mouth hygiene nor shaving would be mandatory for the 14 day mission, the ability of the crew to avail themselves of these measures is expected to have an excellent effect upon morale and performance.

The estimated weight of the food, water, containers and preparation equipment for the intermediate acceptability diet and the water supply system described are listed below:

	Weight (lb)	Volume (cu ft)
Food	130	5
Containers and Shelving	53	1
Water	62	1
Water Containers and Bladder	6	.2
Primary Water Recovery Equipment	10	.6
Urine Water Recovery Equipment	27	1.2
Food Service Equipment	5	.2
Hygiene Equipment	4	.4
	297	9.6

5.5 WASTE

The design criterion for the waste collection and handling subsystem is given in the APOLLO Interim Report. In summary, these criteria call for a light weight, sanitary device designed for male usage, which will operate at 7 psia and in zero gravity, require minimum handling and which is suitable for use by a crew of 3 for 14 days.

After a consideration of various approaches to the problem of waste collection, the approach selected was that developed by the American Machine and Foundry Co. for WADD. This unit generally satisfies all of the above criteria and has the additional advantage of requiring no power for its operation. Feces and urine are collected separately in sealable containers. The urine collection container can be constructed such that transfer to the urine recovery equipment is readily accomplished.

5.5.1 Feces Collection

The Astronaut removes the seat and container caps, unrolls the container, and then inserts the assembly in the seat. During insertion, the orientation of the ring is maintained so that the male end of the pressure line pierces the container and enters the female receptacle in the ring. He straps himself into the seat and operates the hand pump. A partial vacuum is created in the container by exhausting excess air into the cabin through the relief valve, thus sealing the man's buttocks to the container form. The excess air is sucked from the container through the tube, vacuum line and deodorizer and pumped to the relief valve. A schematic of the feces collection subsystem is shown in Figure I-5-4.

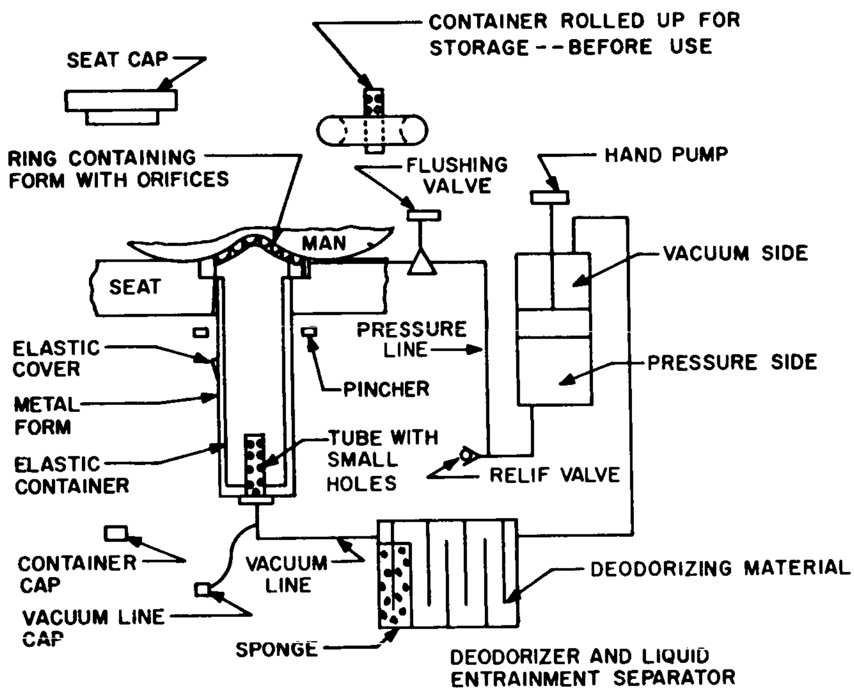


Figure I-5-4. Pneumatic feces collection system

During defecation, the pump is operated as required to maintain the vacuum. When defecation has been completed, the flushing valve is opened and the hand pump operated. Air and gases from the container circulate through the tube, vacuum line, deodorizer, pump, pressure line, form and back into the container through orifices spaced around the container ring. Pumping is continued until the odors have been removed and the jets of air from the small orifices in the ring have forced the feces deep into the container.

The pinchers are then operated, squeezing the container, and sealing the feces in the bottom. The ring is removed from the container and the container top is closed with a spring clip. The vacuum line is disconnected and it and the container are capped. The container is removed from the seat and stored in an appropriate location.

The system has the advantage of not requiring the venting of cabin gas to outer space. Sanitary conditions are easily maintained in that the buttocks are separated from the collection seat by the disposable container and feces never contact the seat or system.

5.5.2 Urine Collection

The collection of urine in the absence of gravity is not as difficult as the collection of feces, since urine is imparted with a velocity when expelled from the penis.

The receptacle for the penis is a self-sealing device. It is formed of rubber and has a receptacle containing a diaphragm to prevent backflow of urine. This seal technique is reliable and can be used under weightless conditions. The receptacle which collects and retains the urine consists of a bellow-type collapsible container. To operate, the bellows is squeezed together and the penis is inserted into the attached receptacle. The crew member then urinates and simultaneously releases the bellows at a rate which still provides a slight vacuum.

The waste collection assembly will be located in the mission module and will be installed with a surrounding curtain for privacy. Its installed weight is estimated at 13 pounds and volume at 1.6 cubic feet.

In the event that an irrepairable cabin decompression has occurred and the crew has retreated to the emergency pressurization capsules they will have lost access to the waste collection device. For this emergency sufficient plastic containers similar to those used for defecation and urination will be provided in each capsule. While defecation in zero "g" without the assistance of the device described above will be less than optimum, the use of simple plastic containers will certainly suffice for this emergency condition.

5.6 PERSONAL EQUIPMENT

5.6.1 Survival Equipment

Equipment suitable to support the life of the APOLLO crew for 72 hours after an unprogrammed landing will be provided within the re-entry vehicle. This equipment will provide sustenance and protection assuming either water or land impact at any spot or environment on earth.

Maximum use of the vehicle can be expected should a ground recovery be made. That is, the capsule can be used as shelter in particularly cold climates or the parachute used as cover in desert type environments. Although the capsule is designed to float, a life raft is provided in the event that damage upon landing results and the vehicle ships water. The raft will be packaged in the parachute housing so that it is readily available after emergency egress. Consideration shall be given to automatic jettisoning and inflating of this raft at parachute release.

Except for those particular items such as the raft and the equipment normally stowed within it which pertain to water survival only, the equipment will be packaged in readily transportable containers in the event that the decision to "walk out" is made.

A list of the survival equipment which will be required in the APOLLO vehicle is presented below:

Item	Weight (lb)	Volume (cu in)
Chap Stick	.03	1.
Insect Repellant (3 reg)	.93	24.
Snake Bite Kit	.23	9.
Sunburn Ointment, can (3 reg)	.48	15.

Item	Weight (lb)	Volume (cu in)
Fishing Kit	.47	15.
Matches, 170/roll	.50	36.
Life raft, 3 men	33.00	4000.
Compass	.32	44.
Manual, Survival	.44	15.
Salt tablets	.09	1.
Universal Clothing Kit (3 reg)	14.25	1050.
Mirror, Signal	.36	8.
Smoke, Signal (3 reg)	1.35	24.
Individual food packet (9 reg)	19.08	360.
Axe	1.69	238.
Knife, Hunting	.28	20.
TOTAL	73.49	5860.

Sixty pounds of water suitable for use after impact is also provided. A description of the source of this water has been presented in the nutrients section above.

5.6.2 Clothing

Since the environment which is created within the vehicle follows the shirt-sleeve flying philosophy, no requirement exists for special protective clothing. Ventilation garments, anti-g suits, anti-exposure garments and the like have no purpose in the APOLLO vehicle. The restraint system is a part of the seat and remains with it obviating the need to wear an integrated harness.

Should head protection be desired by the crew while learning to move about in weightlessness, the helmet provided as a portion of the restraint system will be used. Light cotton undergarments will be provided in sufficient quantity to permit a number of changes consistent with the results obtained during habitability studies. A coverall similar to a summer flying suit will also be provided. This garment will have Velcro patches suitably located on chest, waist, lap and arms as an aid to securing items such as tools, pencils, check lists, navigation aids, etc.

Soft-soled slipper type shoes will be offered for maximum comfort. Velcro material will be bonded to the shoes with mating material placed upon the "floor" of the vehicle. Zero-g flight tests of Velcro bonded to shoes have been extremely satisfactory as an aid to movement and control within a cabin. In addition, this method provides a light "anchor" when a stationary position is desired.

As the design progresses and the crew tasks are more precisely defined additional equipment might be added since integration of tasks, cabin furnishings and clothing is in some cases necessary in order to aid in achieving the maximum possible crew performance.

5.6.3 Recreation Equipment

The requirement for exercise to maintain muscle tone while weightless for prolonged periods has been investigated. The closest analogy which can be drawn is that of extended bedrest by patients of hospitals. Experiments have been performed wherein correlation has been made between effect upon muscle tone and amount of exercise afforded patients. This information indicates that a few minutes of well-programmed activity will prevent all but negligible amounts of muscle atony.

While not completely representative of weightlessness, this information would indicate that ten or fifteen minutes of exercise each day will be sufficient during this mission. The type of activity suggested is that sometimes known as "dynamic tension" where muscles are caused to work against each other through application of opposing forces. The advantage of this method is that no additional equipment is required within the vehicle to perform this exercise.

With regard to other recreation it is felt that this will be individually oriented with respect to each crew member. As an example, literature which might be carried will vary with taste and individual interest. Investigations during habitability and confinements tests expected to be conducted during this program will shed light upon the type of equipment which will be most suitable to the problem. Allowance has been made for the weight of this gear although it cannot be accurately defined at this time.

6.0 Weightlessness, Artificial Gravity and Bioinstrumentation

6.1 PHYSIOLOGICAL EFFECTS OF PROLONGED WEIGHTLESSNESS

6.1.1 Introduction

The early history of the weightlessness problem, as studied principally by means of Keplerian flights in aircraft at the School of Aviation Medicine, Randolph Air Force Base, has been reviewed by Campbell and Gerathewohl. A historical review of studies performed at Holloman Air Force Base is also available. The significant work of Brown at Wright Air Force Development Division (WADD) on human performance capabilities during such flights is well known. Such experiments, although markedly limited in duration of exposure to weightlessness (14 to 45 seconds), have served to disperse the many early uncertainties and fears which surrounded this problem. Whether data gathered during such flights, as well as the data available from short ballistic flights with animals, are applicable to the problem of long-term weightlessness is debatable. In this section, speculation will be made on some of the possible consequences of long-term weightlessness and methods derived for the prevention of untoward effects. The production of artificial gravity by rotation raises additional problems related to stimulation of the semicircular canals. For the APOLLO mission the weightless environment will be a requirement and the evidence at hand suggests that man will be able to function adequately if appropriate precautions are taken. Final decision must await the results of the Mercury manned flights. The Russian finding is in accord with this view.

It is generally assumed that on a space platform, orbiting about the earth without drag from the outside atmosphere, the gravity-free state is realized in the most perfect manner. While this proposition is entirely valid for the time scale and motions involved in physiological processes, it is not rigorously correct from a theoretical standpoint. Schaefer has emphasized that the gravity-free state in an orbiting vehicle of finite dimensions prevails only in its center of gravity. At any other point, the balance of centrifugal force and gravitational attraction is not complete. Minute as this difference is, the consequence may be considerable in terms of crew performance, particularly when free motions of objects on a time scale of several minutes or more are involved.

6.1.2 General Metabolic Effects

There appears to be a reasonably clearcut relation between metabolism and gravity. The metabolic costs of passive standing were first demonstrated by Benedict and Murchhauser in 1915 and the data are shown in Table I-6-I. Tepper and Hellebrandt, in a study of 75 young women, found a metabolic increment between recumbency and passively assumed standing on a tilt table of +5.71 cals/sq m/hr or 16.25 percent. In 31 experiments on 12 healthy young women, Turner, Newton and Haynes found an increment of +5.8 percent at 62 degrees tilt and 19 percent at 90 degrees. The rise in metabolism on passive standing arises presumably because of gravitational stimulation of proprioceptors in the muscles and joints of the lower extremeties and, therefore, an increase in muscle tone.

TABLE I-6-I. METABOLIC EFFECTS OF POSTURE

Condition	O_2 Consumed cc/minute	Energy Output Calories/minute	Energy Output Calories/hour	%
Lying Down	226-242	1.14	68.4	100
Sitting	234-260	1.19	71.4	104.4
Standing at ease	238-239	1.25	75.0	109.6
Moving arms	516	2.53	151.8	222.0

The relation between metabolic rate and postural muscle activity suggests that the metabolism of weightlessness may be closely related to that of recumbency and inactivity. In space craft free motion may be limited, thus reducing exercise. Energy costs for self-propulsion under weightlessness may, on the other hand, be high until coordinated body movements are achieved by practice. In general, if space suits and restraint systems are used and crew tasks include largely monitoring and observation, there may be a need for programmed exercise. For instance, oxygen consumption of personnel in current space suits carrying out ordinary pilot activity over a 12-hour period averaged in one study 300 ccs/min, and in another 390–480 ccs/min, the higher level (480) occurring during pressure suit inflation. No significant differences in oxygen consumption occurred when pilots performed first in summer flying suits and then in uninflated pressure suits. Inflation results in significant increases in metabolism due to the extra work necessary to move the distended joints.

Confinement and inactivity will probably be a characteristic of space flight but it is not possible presently to separate metabolic effects related to inactivity from those due to weightlessness per se. It is instructive, however, to extend the consideration of the life in confinement as a partial analogue of life in a space craft. Studies of inactivity and confinement have been conducted along two lines: (1) the metabolic effects of immobilization and bed rest, and (2) the effects of submersion in water. The most detailed observations have been made under conditions of partial immobilization in a plaster cast.

6.1.3 Immobilization

In a study by Deitrick, Whedon and Shorr four healthy, normal young men were immobilized in bi-valved casts extending from the umbilicus to the toes for six to seven weeks on a constant dietary intake. Under these circumstances a decline of 6.9 percent in metabolic rate occurred. Nitrogen losses averaged 53.6 grams and were associated with a lowered creatine tolerance and a decrease in muscle mass and muscle strength of the immobilized limbs. Calcium losses ranged from 9 to 23.9 grams. The calcium content of the urine was doubled and this, together with the absence of appreciable increase in urine volume, a slight rise in urinary pH, and the failure of urinary citric acid to rise parallel with the increased urinary calcium, favor the precipitation of calcium phosphate in the urinary tract, although this did not occur. A deterioration in the mechanisms essential for adequate circulation in the erect posture was apparent as a tendency to faint on tilt-table tests. Blood volume declined, exercise tolerance decreased and resting pulse rate increased. Complete recovery of metabolic and physiological functions occurred slowly, requiring as much as six weeks in some cases. In the case of the astronaut no such severe effects are expected, but the effects of confinement and restricted activity may be considered additive with those of weightlessness.

6.1.4 Submersion

Water immersion has been suggested as a simulation of weightlessness. The simulation derives from the fact that the relative densities of the human body and water are nearly the same, which almost abolishes the "contact forces" and the resulting internal deformations. The body is still acted upon by "field forces" due to the relative densities of different parts of the body. The simulation is only partial and is complicated by the fact that the body is surrounded by water rather than air, and inertial forces are generated due to the resistance of the water when the subject moves. Although water immersion has been used widely to study spatial orientation, sensory deprivation and acceleration protection, all areas of interest and application for space flight, only recently have such studies been related to the problem of zero g. It is characteristic of these studies that no experimental controls were provided and mechanisms for the prevention of effects not examined.

In a study by Graveline and Balke large nitrogen losses were observed but only relatively minor changes in calcium excretion. Polyuria was marked. Some increase in WBC values occurred and the hematacrit reached 57 on the third day of the seven-day study. Average caloric intake during submersion was 1900 calories. A comparison of pre-and post-immersion work capacity, orthostatic tolerance and g-tolerance was made. All measurements paralleled those expected on the basis of bed rest data but the changes were early and marked. Treadmill tests showed decreased work capacity; tilt-table tests showed a decreased orthostatic tolerance; and centrifuge tests demonstrated a decline in $+G_z$ -tolerance. In the latter case, measurements of $+G_x$ tolerance would have been more desirable if the data are to be applied to space flight.

Graybiel and Clark have also used submersion to assess the effects of prolonged weightlessness. In their experiments, subjects were exposed only 10 hours per day for two weeks, the remainder of the time being at bed rest except for experimental observation periods. Systematic attempts were made to eliminate a number of effects due to sensory depreviation, i.e. someone was with the subject 24 hours a day. Although no decrement in muscular strength and coordination occurred, marked postural hypotension developed during and following the period of immersion.

Because of the reduction of metabolic requirements occasioned by weightlessness, interest has been expressed in sleep requirements for the crew of a space vehicle. McKenzie, Hartman and Graveline have evaluated sleep requirements during immersion and found a reduction in the total sleep requirement, a constriction in the range of sleep states and a progressive improvement in the stability of sleep states. These authors suggest that sleep schedules in space may be shorter than normal, that subjects can easily be aroused to meet emergencies and during the experimental periods observed no effects attributable to sleep loss. The observations of Lilly tend to confirm these findings. Graybiel and Clark, on the other hand, found no sleep changes and attribute such effects to individual differences and motivation, reflecting subject selection. They point out that men on long submarine cruises typically sleep most of the hours when not on duty. Although psychomotor performance may decline during immersion studies, such changes also may be related to motivation. Further studies are needed in this area.

6.1.5 Cardiovascular Effects

It is widely believed that the postural hypotension which develops during prolonged inactivity, whether produced by bed rest or by submersion, reflects a profound change in endovascular reflex responses. The experimental evidence appears, however, to support the view that these effects are due to an alteration of circulating blood volume and a redistribution of blood within the body resulting from recumbency and inactivity.

Stretch receptors and baroreceptors have been known to exist in various parts of the circulatory system. The characteristics of the redisposition of the blood volume accomplished by changes in posture, cuffing of the limbs and obstruction of the vena cava have suggested to several authors that volume receptors might exist in the upper half of the body. Attention was drawn to the neck itself by experiments that suggested that the application of a cuff around the neck might modify salt excretion. However, this has not been confirmed by others.

Recently, considerable attention has been directed to volume receptors within the thorax. Gauer, Henry, Sieker and Wendt demonstrated that negative pressure breathing promoted an increase in urine flow in dogs while positive pressure breathing promoted a decreased flow. These observations have been confirmed in man and it seems reasonably clear that the response is an excretion of water with no primary effect on the rate of excretion of sodium or other solutes. The response is blocked with vasopressin and no increase in urine flow occurs while the subject is under maximum water diuresis and the secretion of anti-diuretic hormone is, presumably, suppressed completely. The anti-diuretic response can be partially or completely inhibited with alcohol. Surtshin et al have demonstrated that the diuresis of negative pressure breathing is not significantly affected by renal denervation. Thus, the efferent arm of this reflex seems to be a diminished supply of anti-diuretic hormone rather than some peripheral neural component.

Henry and his coworkers demonstrated that distention of the left atrium with a balloon resulted in a pronounced diuresis and that the response to negative pressure breathing was either abolished or reduced by section of, or application of cold to, the vagus nerves. They also recorded neural discharges from the vagus nerve and by relating peak activity to events of the cardiac cycle were led to conclude that the receptors primarily respond to stretch rather than to pressure.

Henry and others seemed to have excluded distention of the pulmonary arteries and venous system, except, perhaps, for that portion of the pulmonary vein that lies within the pericardium. Love et al demonstrated that the amplitude of the pulsations of some parts of the intrathoracic vascular system might be the important stimulus. They reported that pulsatile pressure breathing around a mean of zero, or at a positive pressure of 20 mm Hg, promoted an increase in the urine flow, whereas nonpulsatile positive pressure breathing did not.

In assessing the significance of this atrial reflex Henry, Gauer and Sieker reported that changes in blood volume from -30 percent to +30 percent influenced pressure concordantly throughout the circulatory system, and concluded that stretch receptors in the left atrium could, therefore, be influenced by changes in volume. Recent studies by Lewis et al. have demonstrated the extent of the blood shift to the lungs which takes place on recumbency.

Superdiaphragmatic inferior vena cava constriction, which results in an expansion of the low-pressure venous system below the constriction and a depletion of blood above, results in an increase in the secretion of aldosterone. The effect is reversed by a rapid infusion of blood above the constriction. The evidence suggests that a neurohormonal hemodynamic mechanism plays a role in controlling aldosterone secretion. Farrell has suggested that stretch of the right arterial receptors evokes afferent impulses that lead to the inhibition of the release of another hormone from the diecephalon which, in turn, regulates the secretion of aldosterone by the zona glomerulosa of the adrenal gland. This reflex might result in some increased loss of sodium chloride from the body.

It seems possible that the submersion data of Graveline and Balke shown in Table I-6-II may be explained in the following manner: submersion (including the attendant inactivity) results in blood shifts within the venous system, leading to diuresis and natriuresis, perhaps due to the stimulation of stretch receptors in the thorax. There results, over a period of time, a decrease in circulating blood volume, in part due to the diuresis and in part due to a translocation of fluid to the extravascular spaces, the latter accounting for the clinical evidence of plethona reported. These complex changes, resulting in a diminished blood volume, account for the orthostatic hypotension and the diminished g-tolerance. Further studies will be required to confirm such speculation and to define the mechanisms of blood volume alterations which appear to occur during recumbency, submersion, and inactivity. These are the factors which appear likely to occur during weightlessness, and attention must be given to means of preventing their occurrence. Passive exercise by means of an oscillating bed has been shown to considerably modify the metabolic and physiological effects of immobilization. In general, metabolic abnormalities were reduced by approximately one-half, and deterioration of cardivascular postural mechanisms was largely prevented, although the experimental subjects continued to be confined to casts.

Blood shifts to the chest, if they occur during zero G can be reversed by positive pressure breathing. This procedure would also be of value in preventing atelectasis which might conceivably occur in higher oxygens atmospheres. Procedures and equipment for several sessions of positive pressure breathing per day for each member of the crew should be considered.

TABLE I-6-II. BIOCHEMICAL ANALYSIS OF URINE SAMPLES

	Urine Volume cc/24 hours	Na	K	Ca	Cl	PO4
				mEq/24 hours		
Control	1170	251.7	81.1	10.8	74.9	49.2
Day 1	3200	549.5	83.1	11.8	137.1	35.1
Day 2	2900	345.0	79.4	7.8	77.6	45.1
Day 3	3000	221.6	103.6	12.9	55.1	61.7
Day 4	1750	175.7	80.6	7.5	39.2	51.1
Day 5	2100	131.1	70.6	8.7	43.6	42.3
Day 6	1800	192.3	74.3	10.4	50.1	45.4
Day 7	1550	307.5	40.7	10.9	66.7	39.3

6.1.6 Muscular Atrophy

In the submersion study of Graybiel and Clark, muscle strength was fully maintained and endurance, as measured on the treadmill, was reduced in only two subjects. These results are at variance with data obtained during immobilization in plaster casts, and that obtained by Graveline and Balke. In discussing their findings, Graybiel and Clark cite the experimental studies of Mueller and Hettinger and those of Rose et al. on the importance of daily muscular training. Muscular exercise constitutes a training stimulus to a muscle if it exceeds one-third the maximum muscular potential. Much less than this value is required to prevent atrophy. If exercise exceeds the one-third threshold value, an increase in muscle strength will result. Daily exercise that leads to a rapid increase in muscle strength will show an equally rapid decline on cessation of exercise, while a high muscular potential (50 to 80 percent over the initial strength) could be maintained or even slightly increased by once-a-week maximum contractions. Reductions in muscle strength occur if exercise is limited to once every three weeks. Muscle strength is better maintained by slow versus rapid development.

In Graybiel and Clark's study, apparently the small amount of muscular activity involved in getting in and out of the water, moving in bed, etc., was sufficient to meet the required criteria indicated by Mueller and Hettinger. In any event, the muscular movements were far less in Graybiel's experiment than would be experienced under zero g conditions, even if the persons were confined to a seat or couch. However, under zero g, certain antigravity muscles might not be called into play by crew duties.

It has been proposed that exposure to prolonged weightlessness might result in the deterioration of the cardiac musculature and ultimately in heart failure. The suggested mechanism is a pronounced decrease in cardiac work. Although a decrease in cardiac rate may occur during zero g associated with a more general decrease in body metabolism, the major elements of cardiac work should be unaltered. The principal contributions to cardiac work occur in overcoming the inertia of the blood, the production of flow against viscous forces and the distention of the vascular system, none of which will be altered by zero g. The absence of gravity should not alter the hydrostatic load on the heart. Experiments during acceleration on centrifuges have demonstrated an hydrostatic balance point at the level of the heart in the arterial system (Wood, Lawton). A similar balance point occurs in the cerebral spinal fluid (Rushmer).

6.1.7 Bone Demineralization

Under conditions of complete immobilization of the lower extremeties by means of plaster casts, bone demineralization is pronounced. Maximum urinary excretion of calcium occurs about the third week and taken together with other urinary changes suggests that calculus formation might occur. In submersion experiments, such as those at Brooks Air Force Base, no disorder of calcium excretion was observed in the seven-day experimental period. Thus, calcium balance depends on the amount of immobilization and does not appear to be reflexly regulated as is the case with sodium.

The disposition and mobilization of calcium depends upon the bone intramedullary pressure. Deposition is enhanced by bone compression. Clinical observations indicate that extremeties placed in traction (tension) do not show normal bone growth and calcium deposition. The relation between compression and calcium deposition has been demonstrated also in tissue culture with isolated bone cells. No evidence exists that in clinical cases disturbance in bone blood flow accounts for the demineralization which occurs at bed rest.

Over the short exposures of the APOLLO mission, bone demineralization is unlikely to be a problem unless the crew is severly restrained. If this proves to be a problem, exercises of the "Charles Atlas" type should result in adequate deposition. Individuals who consistently use such exercises show marked increases in bone size due to calcium deposition, as well as the familiar muscular hypertrophy for which the exercise is undertaken.

6.2 ORIENTATION, ROTATION AND ARTIFICIAL GRAVITY

6.2.1 Otolith Functions

Submersion in water has been used to study orientation. Under these conditions visual cues, tactile cues, kinesthetic cues, temperature cues, and buoyancy cues may be eliminated to a large extent by appropriate experimental procedures and orientation to the vertical may be indicative of perception via the labyrinthine senses. Human subjects may very easily be disoriented with respect to the direction in which they are facing by rotation about a vertical axis. Such disorientation is attributed to the effects of angular acceleration and deceleration on the semicircular canals. It does not follow that they can be as easily disoriented with respect to the direction of the vertical by rotation about a horizontal axis. Rotation about a horizontal axis in a vertically oriented gravitational field results in a characteristic changing pattern of stimulation of the utricles on each rotation. Thus, in addition to the disorienting effect of semicircular canal stimulation, there is another source of stimulation which could, conceivably, enable the retention of orientation with respect to the vertical.

In 1942, Adrian demonstrated a prolonged discharge in the afferent neural connections of the utricles following changes in linear acceleration acting on the body. Presumably the utricles signal changes in the orientation and/or magnitude of linear acceleration forces. An individual immersed in liquid may therefore be expected to have some basis for orientation in the absence of the usual visual, tactile and other cues. This is not to say that the utricles should be expected to provide him with a continuously available sense of reference for the gravitional vertical, but following a few simple changes in position of the head, the pattern of change of utricular stimulation may afford a cue as to the orientation of the vertical or "which way is up." The study of utricle function thus has significance for zero g space flight.

In spite of its disadvantages, immersion as a method of stimulating zero g is believed to have some utility. Muller has suggested that the response of subjects while immersed might be studied by means of a liquid-filled cylindrical capsule which may be rotated about a horizontal axis to eliminate any constant vertical reference for the subject who is held in a fixed position inside the rotating capsule. With a fixed visual display and, if head movement is prevented, some rate of rotation might be found at which there would be a ''fusion'' of the stimulating effects of rotation on the utricle. At this rate there would be no available vertical reference for the observer. Rotation may not be necessary, if the subject is restrained within the tank, particularly when he is prevented from making any head movements. The possibility of utricular cues as to the orientation of the vertical in the absence of any changes in utricular stimulation is highly questionable. This general problem is under study by Levine.

Recently, the utricle has been studied under weightlessness as well as submersion. King has investigated postural orientation by tilting during weightlessness in decerebrate pigeons, and Johnson has identified the importance of the utricle for the perception of gravity in lower animals during Keplerian flights. Shock has found impaired orientation to the vertical during submersion and has raised the question of the need for artifical gravity during space flight. Whiteside and Chambers have tested orientation as measured by pointing during submersion in water up to the neck. Some loss of directional sense was found in this situation even though the utricular sense was not diminished. The loss was attributed to the altered muscle balance, absence of visual information and reduced proprioceptive cues. The data of Pearson and Hauty, obtained by lateral tilt in a dark room, in which the subject returns himself to the preconceived vertical, suggests that some proprioceptive learning may occur.

Many attempts have been made to measure the threshold of the utricles for changes in resultant linear acceleration. Reported threshold values range from 0.000344 g to 0.010 g or higher. A number of determinations were made on tilt tables in which the subject determined the minimal detectable change in table orientation when the angular accelerations of the table were sufficiently low to avoid stimulation of the semicircular canals. Such experiments have little value because of the number of postural cues available, i.e. tactile stimuli from restraining straps, noise, and mechanical irregularities of table motion. More reliable data were obtained by Graybiel and

Patterson in terms of the oculogravic illusion. They reported a threshold of 0.000344 g for subjects in the sitting position and 0.00203 g for subjects lying on their sides.

In 1928 Quix reported the presence of a "blind spot" in the utricular sense for subjects in a supine position with the head depressed. Knight attempted to repeat the experiment of Quix under water. Although he found higher thresholds with the head oriented downward, he was unable to obtain clearcut, quantitative results.

Stigler in 1912 attempted to measure the orientation of swimmers after they had been rotated on a bar while completely submerged. Eyes and ears were covered to eliminate visual and auditory cues. They were instructed to point in an upward direction upon termination of the rotation. Apparently they were seldom able to do this accurately; the experiment was reported as unpleasant and anxiety-provoking and was terminated because of the subjects' inability to hold their breath long enough for adequate observations.

Recently Brown has repeated these observations using the U. S. Navy Diving Tank at New London. Three specific experimental questions were posed: (1) when submerged in a tank at a point near neutral bouyancy can a man, placed in some random orientation after several disorientating turns by the experimenter, point correctly in the direction of the vertical without any gross movements of the head? (2) If he is unable to point directly in the direction of the vertical, can he, after some exploratory head movements, point in the vertical direction? (3) How accurately can he orient his entire body to the vertical and swim in the direction of the surface?

The experimental procedure was refined in the following particulars: little if any temperature gradient was present in the tank, (2) experiments were performed at neutral bouyancy, (3) experienced divers were used as subjects who could hold their breath sufficiently long for the experimental procedure without exhaling and producing bubbles. Visual cues were eliminated by an opaque face mask. In spite of the probable residual cues which existed in addition to those provided by the utricles, (i.e. difference in density of various parts of the body, slight positive or negative bouyancy), the relation of accuracy of orientation to the terminal position at the end of rotation and the improvement of orientation following head movement were interpreted to indicate that the utricular sense was prominent in this situation. Trials which were terminated in the head-backward, face-upward position or the head-downward position resulted in greater deviation from correct orientation than trials terminating with the head up or forward and may be compared with the results of earlier experiments.

The existence of a utricular sense has been questioned. Few present-day investigators doubt it, but it has probably never been measured acting all by itself. Gray has pointed out that its normal function is in conjunction with the semicircular canals and it probably should not be considered an independent sensory component.

Recent studies by Johnson and Taylor have directly attacked the problem of separating otolithic and semicircular canal function. Using a counter rotating turntable mounted on a second turntable revolution without rotation may be produced. Although the subject is being rotated on one turntable, the counter rotation of the second results in the subject always facing in the same direction. Using glass models of the equilibrium organs it has been demonstrated that only linear accelerations are produced during the exposure and studies of tilting of after images are attributable to otolithic stimulation alone.

6.2.2 Semicircular Canal Phenomena

When a subject is passively rotated in one plane and turns his head in another, a subjective sensation of rotation in a plane approximately orthogonal to the other two is produced. The total sensation is complex. For instance, the sensations of angular speed and displacement may be discordant, the plane of apparent body rotation may shift, visual illusions may be perceived, and symptoms referable to many bodily systems produced.

Of particular interest has been the so-called oculogyral illusion which may be defined as the apparent motion of objects in the visual field having its genesis in stimulation of the semicircular canals by Coriolis accelerations. Visual illusions related to stimulation of the otolith organs are called oculogravic illusions. The oculogyral illusion has been used as an experimental measure of semicircular canal stimulation. Another experimental measure is vestibular nystagmus (oscillatory motions of the eyes) produced under similar circumstances. Both measures may be used to follow the time course of the adaption to vestibular stimulation. The oculogyral illusion is not related perfectly to visual nystagmus but corresponds well under certain experimental conditions.

Recently, Hallpike and Hood have developed quantitative evidence of the relation between nystagmus and the oculogyral illusion and have offered evidence to substantiate Steinhausen's general theory of the cupula mechanism and the physical constants of the system assigned to it by Van Egmond and his co-workers.

Steinhausen was able to establish that the cupula is hinged upon the crista and fills the lumen of the ampulla. Following displacement the cupula returns slowly to the rest position under the influence of its own restoring force. The characterization of the cupula as a torsion pendulum has made possible the calculation of its responses by means of appropriate differential equations. This was carried out by Van Egmond and it is now possible to explain the lengthy after-period of post-rotational sensation in terms of the slow return of the cupula to its rest position (Figure I-6-1).

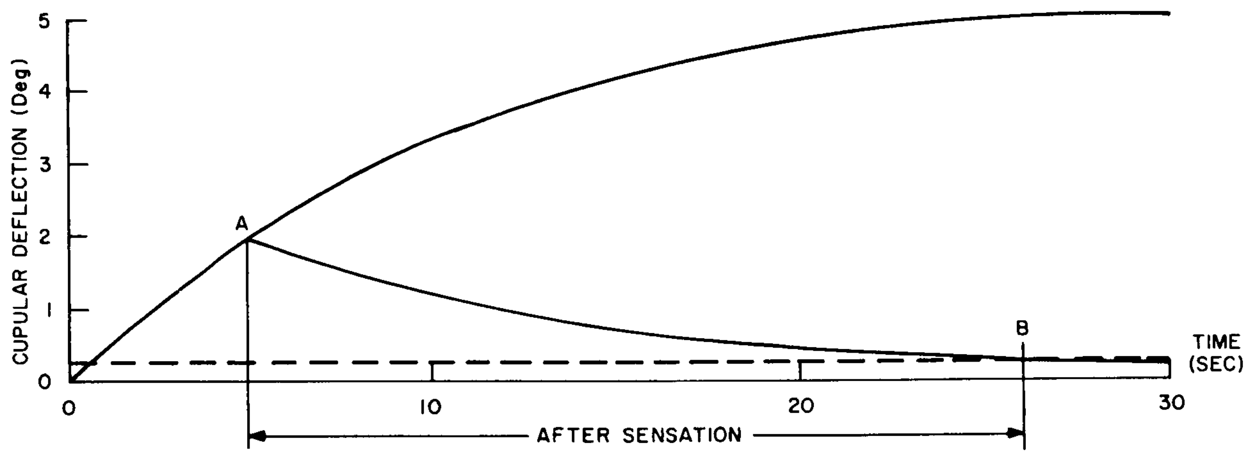


Figure I-6-1. Theoretical time course of cupular deflection in response to angular acceleration of 5° Sec^{-2}: AB, return of cupula to rest position from a deflection of 2°.

It has become common knowledge that the physiological responses of the cupula to sustained angular accelerations exhibit no systematic relationship, useful in the physiological sense, with the various physical quanta, i.e., angular velocity, angular acceleration, etc., inherent in the stimulus. In Figure I-6-2A the manner in which the maximal cupula deflections attainable in response to prolonged accelerations increase with their magnitude is shown. In Figure I-6-2B it is seen that in the earlier course of application of these accelerations, the moment-to-moment values of the cupula deflections exhibit no constant relationship either to the magnitude of the accelerations being applied or to the angular velocity attained. This physiological "irrelevance" of the cupular responses to constant accelerations of this kind is usually referred to the unphysiological nature of the stimuli which lie outside the range of motions normally experienced by the head. By contrast, the responses of the cupula, under normal impulsive conditions of head movement, exhibit a much closer relationship to the physical characteristics of the motional stimulus, in particular its angular velocity. It appears that unless a period of constant velocity intervenes between accelerative and decelerative phases, then the cupula deflection is at all times proportional to the velocity of head movement. In general, the time-velocity curve of a normal impulsive head movement is accurately reproduced by the time-deflection curve of the cupula movement to which it gives rise.

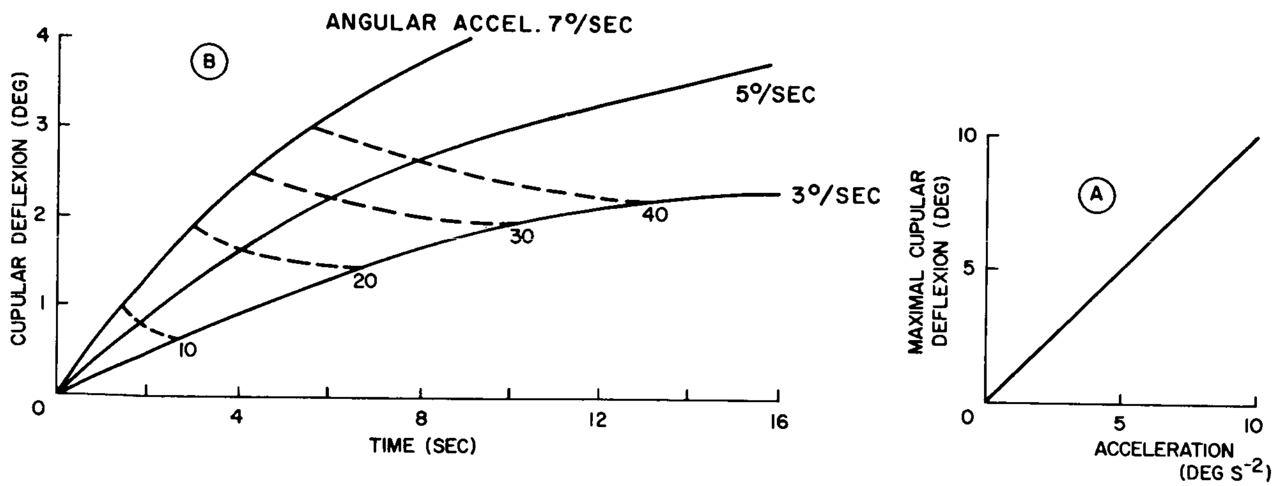


Figure I-6-2. At left, time course of cupular deflection in response to angular accelerations of 3, 5, and 7° Sec^{-2}. Equal velocity contours at 10, 20, 30 and 40° Sec^{-1}. At right, variation of maximal cupular deflection with magnitude of acceleration.

6.2.3 The Slow Rotation Room

Recently, the problem of the tolerance of human subjects to prolonged residence in a rotating environment has been studied in partial simulation of the environment which might be encountered in a rotating space station. A nearly circular, windowless room has been constructed around the center post of the Pensacola human centrifuge. The room is 15 feet in diameter and 7 feet high and may be driven at constant velocity (± 2.5 percent) at rates from 1.7 to 10 rpm.

In such an environment, random or planned head movements stimulate the semi-circular canals as the result of the angular velocity of the head motion itself and of changes in the orientation of the several canals to the plane of rotation of the room. If the head is displaced parallel to the axis of rotation, or laterally in the same plane of rotation, no stimulation occurs. Stimulation results when the head is turned about an axis parallel to the axis of rotation. Turning the head about any other axis results in a complex pattern of stimulation resulting from changes in the direction of the force with respect to the several canals. The magnitude of the effects depend on the specific direction and velocity of head motion and the velocity of the room.

While it is theoretically possible to move the head in such an environment without stimulating the semicircular canals, practically this does not occur. If the head is fixed, the otolith organs are subjected to a constant force which is the resultant of the radial and gravitational forces. However, its direction deviates from the usual gravitational direction giving rise to the oculogravic illusion. Rotation of the head about an axis parallel to the room axis alters the direction but not the magnitude of the resultant force on the otolith organs.

Studies performed in the rotating room to date support the notion that prolonged constant rotation per se, within the range studied (1.71 - 10 rpm) does not interfere with task performance. The dominant stimulus is the random bizarre stimulation of the semicircular canals resulting from head motion. The most prominent change in performance was in motivation towards tasks. Motivation was markedly affected by the occurrence of motion sickness (canal sickness) in the subjects. Sickness resulted in voluntary limitations of head motion and in excessive sleeping. Decrements in walking and body sway tests were substantial during and immediately following rotation. Two subjects showed decrements and subsequent adaptation on the arithmetic test. No decrements were observed in a number of tests of muscle strength and coordination including eye-hand coordination.

6.2.4 Adaptation To Rotation

Whether man can adapt to the subjective and autonomic manifestations produced by head motion in a rotating environment is a subject of some significance for the case of rotating space craft. Observations by Graybiel and his co-workers in the slow rotation room suggest that such adaptation may occur. By the use of a tilt-chair the time course of adaptation to the oculogyral illusion was followed. Subjects were selected to give a wide range of variations. In subjects least susceptible to canal sickness, adaptation occurred in as little time as 16 hours and the illusion could only be perceived at 10 rpm and then only during active head movements. The most sensitive subject perceived the illusion until the last test period at 5.4 rpm and was not tested at 10 rpm.

The results of these experiments indicate considerable capacity in some subjects for habitation to an unusual sensory input. From the practical point of view, it is worth knowing that this illusion and the disturbing autonomic reactions which initially accompany it are substantially reduced and in some subjects disappear during the course of 64 hours of almost continual rotation. As indicated by the oculogyral illusion, most of the decline occurs within the first 16 hours. Following cessation of rotation, illusions are produced by head rotation in the opposite direction. Symptoms may be more severe but are shorter in duration. Although greater responses are produced by active rather than passive movements, it is not clear whether this is due to the greater velocities of head motion, the influence of cervical reflexes, or because of the voluntary nature of the act.

If rotation is to be used in space as a means of producing artificial gravity, much more needs to be known concerning man's response to this environment. The current data suggest wide individual difference among subjects which raises the question of selection and training of the crew. Kraus has reviewed this problem and has suggested tests for selection. On the basis of his data, as well as that of Graybiel and others it would appear that something can be accomplished in this direction. This matter is further discussed under Selection and Training Chapter II, Section 4.0 and 5.0 of this Volume.

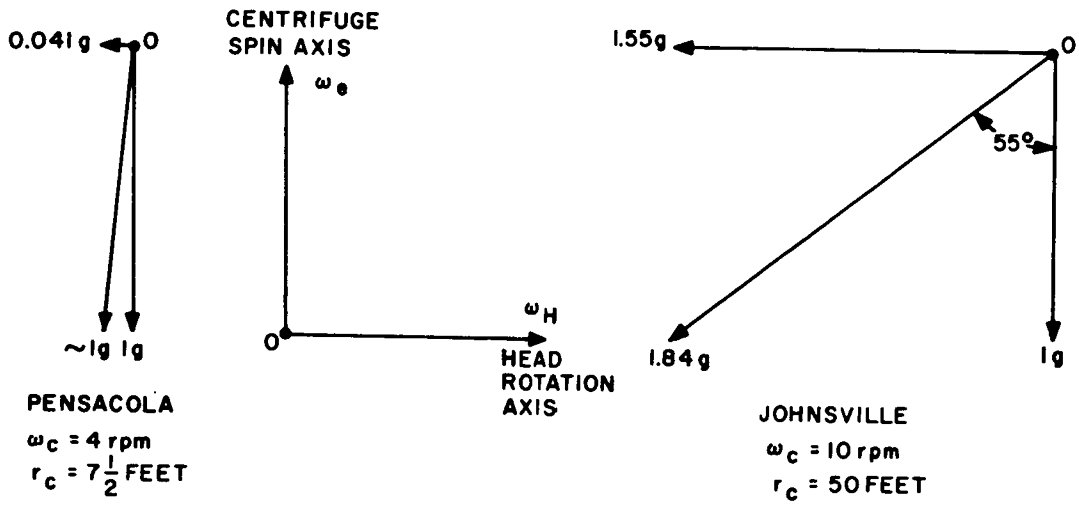
In addition to measures of performance, nystagmus, or visual illusions, other areas of recent study may have a bearing. For instance, Benson has studied reflex movements in response to labyrinthine stimuli, particularly the ankle jerk; and Taylor, Johnson and Sellers have studied cardiovascular changes associated with vestibular stimulation. Something may be learned from a study of figure skaters. Observations on motion sickness indicate a strong psychological element. If selection and training do not yield a satisfactory solution, the use of drugs may ultimately provide a way of extending human tolerance.

6.2.5 Artifical Gravity

For design purposes the following conclusion may be drawn:

1. In the studies conducted at Pensacola, the adaption of the test subjects to the rotation occurred and improved with time of exposure. It was found that, as the rotation rate increased, adaption took longer and fewer subjects could adapt completely. At the lower rotation rates, following adaptation, normal head movements did not produce symptoms, but as the rotation rate increased, the voluntary head movements tended to become more limited. This cross-over or boundary is reported to be between 3.82 rpm and 5.44 rpm or approximately four to five rpm.
2. At Johnsville tests were conducted at a fixed rotation rate of one rad/sec or approximately 10 rpm and fixed radius of 50 feet. It was found in these studies that the threshold for illusions was 0.06 rad^2/sec^2. This is the vector product of the head angular velocity and the centrifuge angular velocity. Head rotations of 0.6 rad/sec in this environment were found to produce nauseating effects. It was noted that the sensation of angular illusions became less distinct as time increased (adaptation), but the nauseating effects remained.

The comparison between the physiological disturbances and the force fields for the two methods of study cited is necessary in order to define limits of angular velocity (spin rate) tolerated by humans. They compare as follows:



It may be concluded that the fields in which the tests were conducted were different in magnitude and direction for each case. Very few subjects have been involved. It cannot be concluded presently whether the test conditions have a direct influence on the test results until additional test data is made available. The tests results from each source have been taken without regard to test method and a determination made of their mutual restriction. For instance, the angular velocity is set at four rpm and if 0.06 rad^2/sec^2 is set as the illusion threshold, the following relationship results.

$$\text{Threshold (Coriolis Force)} = \omega_v \times (\omega_h \times R_h)$$

where ω_v = angular velocity of the vehicle

ω_h = angular velocity of the man's head

R_h = O = radius of man's head

then $0.06\ rad^2/sec^2 = (0.42\ rad/sec) \times \omega_h$

When the head rotation is in a plane perpendicular to vehicle rotation, this reduces to

$$\omega_h = \frac{0.06}{0.42} \text{ rad/sec} = 0.14 \text{ rad/sec}$$

Thus, at the selected rotation rate of 4 rpm, the head must rotate at a very low rate. The calculated rate is below the nausea limit set by Johnsville, 0.6 rad/sec. It is concluded, therefore, that a rotation rate of four rpm is tolerable, but that head rotation rate will be quite restrictive. No data are available on head rotation rates for performance by man confined both by the vehicle and by a space suit.

6.2.4.1 G—LEVELS

For the selection of a g level which may be required for vehicular design, the maximum value can be defined in terms of fatigue of both the vehicle construction and the man. A minimum value may be assigned by the level at which the force field becomes too small to be useful in helping the man feel comfortable or in the performance of his duties. Tests conducted at Mayo Clinic on the human centrifuge showed that, above one g, an accumulative degradation in performance occurred; therefore, one g appears to be an acceptable upper limit. The lower g limit is harder to define because it can also depend on many factors external to the man (effects on the vehicle). Hence, only a preliminary estimate can be made and 0.01 g is considered to be the lowest g value which will prove of any benefit to a human.

6.2.4.2 G—GRADIENT

The variance in the g level between the head and feet of a man in the cabin of a rotating space vehicle may be of concern in defining a comfortable living environment for the man. However, no quantitative information is available concerning the effects of such a g-gradient on pilot performance and comfort. It is believed that this difference in g-level should be held to some low value, perhaps 15 percent as suggested by Payne. In defining this variance in terms of the parameters ω_v and V_v it is found that

g-level		
feet	$\omega_v^2 r_v$	ω_v = angular velocity of the vehicle
head	$\omega_v^2 (r_v - h)$	r_v = radius of rotation
		h = height of the man

and the gradient becomes the difference in the head-to-feet g level divided by the total g at the rim of the vehicle. Thus,

$$\frac{\omega_r^2 - \omega^2\ (r-h)}{\omega_r^2} = \frac{h}{r}$$

For a nominal height of six feet, the value of this ratio becomes simply 6/r. It is apparent that the g-gradient varies inversely as the radius, and a value of 15 percent implies that the radius should be greater than 40 feet.

6.2.4.3 MOBILITY RESTRICTIONS

There are two factors that have a direct influence on man's ability to move comfortably (translation more than rotation) in a rotating space vehicle system. The first is the ratio of the Coriolis force to the centrifugal force, and the second is the rim (floor) velocity.

The ratio of the Coriolis force to the centrifugal force is derived as follows:

$$\frac{\text{Coriolis force}}{\text{Centrifugal force}} = \frac{2m(\omega_v x v_m)}{m\omega_v x(\omega x r v)}$$

m = mass of man

ω_v = angular velocity of vehicle

r_v = radius of rotation

v_m = velocity of man relative to vehicle

If the velocity of man is considered perpendicular to the plane of rotation and the radius of rotation is perpendicular to the axis of rotation, this relationship reduces to $\frac{2\,v_m}{r_v\omega_v}$. Using this relationship, it is noted that the ratio varies directly with the velocity of movement and inversely with the radius of rotation or spin rate.

Thus, if this ratio increases, two distinct effects on the man can occur: (1) the total weight of the man will either increase or decrease, and/or (2) the subjective "down" direction for the man will be altered as the result of visual illusions. The latter of these effects occurs when movement is in a radial plane (between floor and ceiling of the space vehicle) and the former when movement is along the floor of the space vehicle. Therefore, it has been suggested by Kramer and Bajers that this ratio be kept to a low value. It is considered that a limit of 25 percent, calculated using a velocity of three feet/second, is adequate for this mission since the movements of man will probably be restrained to a magnitude below this value.

The problem of the Coriolis Force becoming greater than Centrifugal Force can be quite easily illustrated with the use of Figure I-6-3. For example the lowest point of the bounded area gives

$$\omega = 4\text{rpm}$$

$$r = 57.5 \text{ feet}$$

$$ng = 0.3g$$

The ratio of Coriolis Force to Centrifugal Force can be represented by

$$\tau = \frac{\text{Coriolis}}{\text{Centrifugal}} = \frac{2\ \omega_v \times v}{\omega_v \times (\omega_v \times v)}$$

where:

$\bar{\omega}_v$ = angular velocity of the vehicle

$\bar{v}$ = velocity of the man

Reduce this expression to

$$\tau = \frac{2\ v}{r\ \omega}$$

Then set $\bar{v}$ = 3 feet/sec and r = 57.5 feet, which gives

$$\tau = \frac{6}{57.5\ \omega} = \frac{.105}{\omega}$$

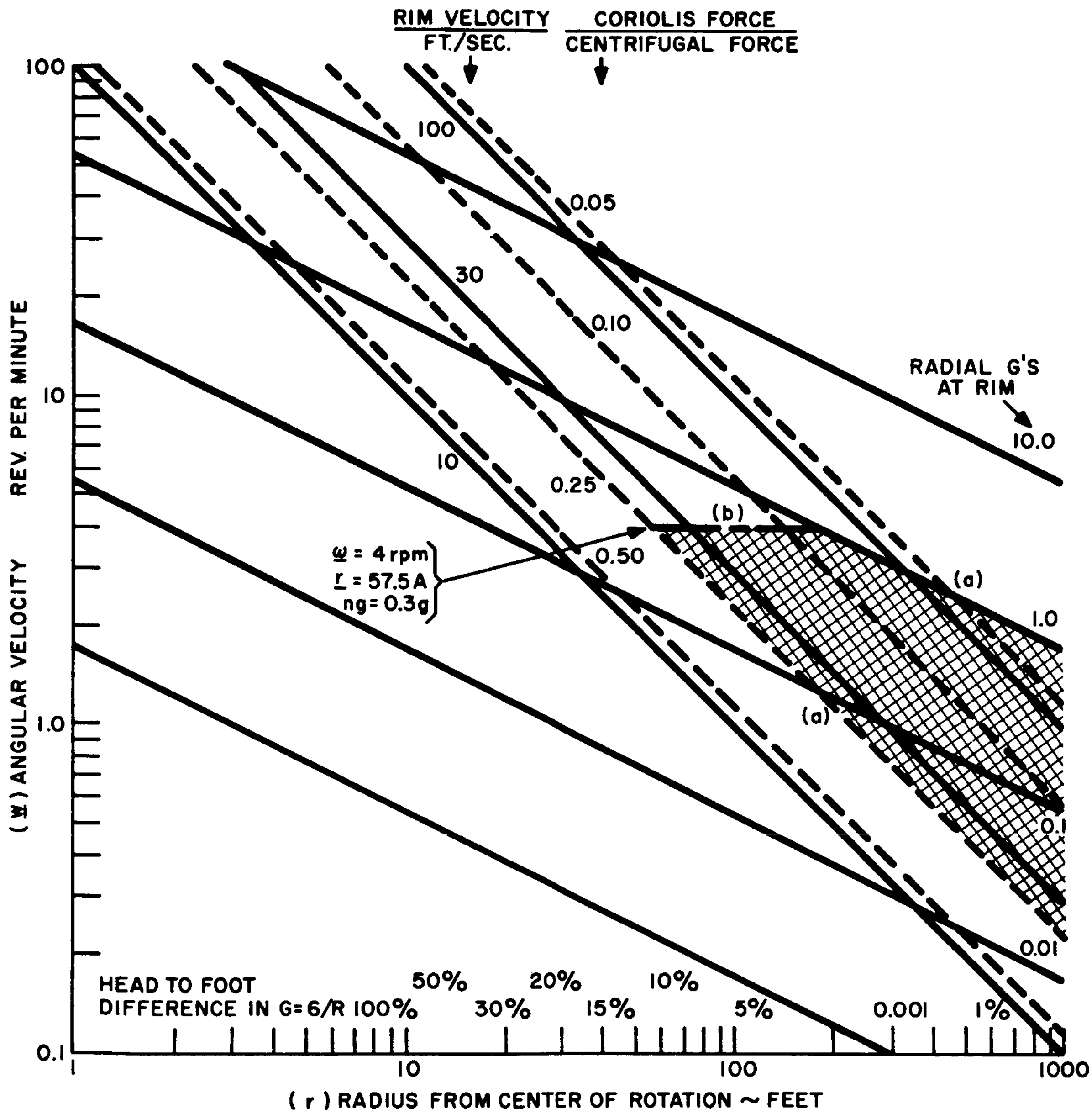
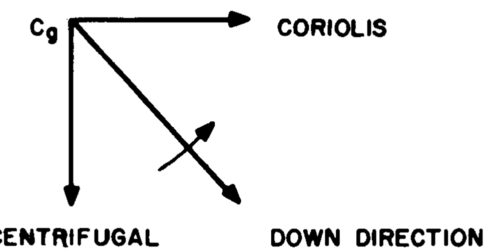


Figure I-6-3. Artificial gravity parameters

Since ω (angular velocity) is in the denominator of the expression it is possible for this expression to become greater than one as the angular velocity becomes reduced in magnitude. In this case the angular velocity would have to be reduced below.

$$\omega_v = .105 \text{ rad/sec} \approx 1 \text{ rpm}$$

If one moves down the constant radius line of 57.5 feet (Figure I-6-3), which in effect is reducing the angular velocity to a value less than 1 rpm (ng < 0.1g), the Coriolis Force begins to exceed the Centrifugal Force which is highly undesirable because



the down direction for the man will rotate markedly.

The second factor, rim velocity, is important in defining the capability of man to walk either with or against the direction of rotation. The Coriolis force acts during walking and the ratio discussed previously must be included. Hence, it is possible to define a rim velocity suitable for walking by the following procedure:

$$\text{Radial acceleration felt by man} = \frac{(V_v \pm V_m)^2}{\tau_v} \pm 2\omega_v V_m$$

V_v = rim velocity

V_m = velocity of man

ω_r = spin rate

R_r = radius of rotation

It is reasonable to assume that the man is carried along with the vehicle at the velocity V_v and when he walks with the rotation his velocity becomes $V_v + V_m$, and when he walks against the rotation, his velocity becomes $V_v - V_m$. Since a man walking against the direction of rotation will experience a diminished g-loading from both the centrifugal and Coriolis forces, it is mainly this limit that is of the most concern. It is believed that an extreme limit of radial acceleration experienced by a man walking against the rotation to a man standing still can be set at 0.5, and this can be expressed by

$$\frac{\dfrac{(V_v - V_m)^2}{r_v} - 2\omega_v V_m}{V_v^2/r_v} \geq 0.5$$

This can be reduced to the form

$$\frac{(V_v - V_m)^2}{V_v^2} - \frac{2\,V_m}{r_v\,\omega_v} - \geq 0.5$$

and it is noted that $2V_m/R_r\,\omega_v$ is the Coriolis force to centrifugal force factor and can be set to the value chosen, 25 percent, so that the equation becomes

$$\frac{(V_v - V_m)^2}{V_v^2} \geq 0.75 \text{ or } V_v \geq 7.7\ V_m$$

This shows that the rim velocity should exceed the velocity of man's walking by at least 7.7 times. For the velocity of three feet/second selected previously, this means the rim velocity should be greater than 23 feet/second to provide the man with comfortable walking conditions.

6.2.4.4 DESIGN PARAMETERS

Based on the foregoing analysis, a preliminary design envelope of acceptable combinations of rotation rate and radius for manned space vehicle systems is given in Figure I-6-3 as a modification of a presentation by Dole.

The limits of the preliminary design envelope are, specifically,

(a) maximum simulated gravity field of one g

(b) maximum angular velocity of four rpm, and

(c) maximum Coriolis force to centrifugal force ratio of 25 percent

These limits have guided the design analysis carried out in Volume VII for a typical artificial gravity system.

6.2.5 Conclusion

From the foregoing, it may be concluded that, at present, there is insufficient information upon which to base an intelligent decision as to whether man can tolerate, or perhaps thrive, in the weightless state, or whether the generation of a gravitational field will be a requirement for his well-being. It is valuable in this connection to consider the APOLLO mission as a whole and the impact of an artificial g environment on its conduct.

Following the launch accelerations, the crew must pass through a weightless phase. Studies by von Beckh have indicated that such transients from acceleration fields to the weightless state may be associated with disorientation. Sometime after entering the weightless state, the craft is set into rotation. During this early phase of the mission to the Moon, critical guidance corrections must be made or the decision to abort taken. Because of the effect of rotation on the precision of the guidance and control system and the requirement to make corrections at a maximum distance from the Moon, at least two to three hours of weightlessness may be required. It is supposed that the Mercury mission will establish that at least 24 hours exposure to weightlessness will be tolerable without undertaking special precautions. Single orbital flight is tolerable according to Russian reports. Probably at least once during lunar voyage rotation may be stopped to carry out path correction. Some course correction schedules may call for as many as four to five derotations. Rotation must again be stopped in order to insert the craft into lunar orbit. Visual observation of the lunar surface by the crew may require a stable, nonrotating platform as may the use of other lunar observation instruments. The illuminated surface of the Moon will be visible for only one hour or less during the three to five hour orbital period. Exit from lunar orbit, mid-course corrections, and terminal guidance all may require cessation of rotation.

It is thus apparent that, depending upon the precision of the guidance and control system and the requirements of the mission, as few as four or as many as a dozen transients between weightlessness and exposure to gravity may be required. In the worst case of 12 transients (poorest guidance and control) periods of exposure to either weightlessness or artificial gravity will rarely exceed one day, a time equivalent to the adaptation period of the most rotation insensitive individuals. On the other hand, the mission under the best circumstances will always require some exposure to the weightless state.

The structural sophistication required to successfully execute a rotating vehicle is considerable. Rotation rates and radii necessary to achieve one g appear unrealistic. If less than one g is achieved, problems of both weightlessness and rotation are likely to appear. It may be argued that the effects of subgravity differ from those of pure weightlessness only in the time course with which they develop. Thus, for subgravity, similar precautions, such as special exercies, may be required.

Subgravity would result in the solution of a number of mechanical problems. Eating, the handling of body waste, the act micturition, the presence of convection, the settling of dust and dirt, and improved equipment reliability, especially those components whose function may be compromised at zero g, are among the advantages. Crew spaces, displays and controls, and the mechanisms by which the crew moves about and does work may be more conventional. However, problems of crew selection and training are likely to be unaltered because weightlessness remains a part of the normal mission.

An over-all view of the APOLLO mission indicates that the problem must be explored further. At least, a firm decision cannot be made until the results of the Mercury flights are known. Two courses of action appear open. First, to proceed, using weightlessness as a design requirement, in gradually increasing steps until sufficient data is generated in the orbital phase to reach a decision, or second, to probe the problem with biosatellites. The latter course would appear to be least expensive and the most rapid way to obtain the required answers. The present work is based on the premise that the combined effects of weightlessness and inactivity can be successfully prevented by adequate and appropriate exercise and that man can perform his role in the APOLLO mission under weightlessness without decrement.

6.3 BIOINSTRUMENTATION

6.3.1 Introduction

Man's exploration of space may proceed either by gradual progression to longer flights at greater altitudes by man himself, or by larger increments of exploration using animal subjects. Both modes of exploration have their adherents. The former case is exemplified by the X-15, the Mercury and the current Russian Programs. In this case gradual extension of the period of weightlessness will be achieved one step at a time and the significant factors of weightlessness, equipment reliability, and safety will be explored simultaneously. The large-increment approach is exemplified by the Laika experiment. Here, advanced information may be obtained, and this may contribute notably to current manned space vehicle programs and provide data for extrapolation to future vehicles.

The exploration of space with biological specimens must involve long lead times. Suitable biological control studies must be generated on the ground. Biological techniques and hardware must be adapted to space flight; this must include detailed environmental qualifications of components and systems. Restrictions on weight and space are likely to affect hardware development, particularly if biological experiments are performed on a "piggy-back" basis.

Because the design criteria of future space vehicles depend so strongly upon adequate experimental biological data which can be obtained only in space, the early acquisition of such data appears mandatory for realistic planning of manned orbital and lunar missions.

6.3 2 Experimental Rationale

Two sorts of knowledge are required for the design of space experiments. The design depends upon taking the best of the current knowledge of human function as derived from physiology and psychology, developed by animal experiments as well as human experiments not performed in space, and the best current knowledge of conditions in space, based upon experiments in space not involving humans but to a certain extent animals. These two pieces of knowledge may then be wedded together to establish hypothetical anticipated performances and thus reaquirements for man in space.

As the result of such an analysis areas may be identified where performance requirements can be anticipated with reasonable assurance. Where this is true, design requirements for manned space vehicles may be established. Other areas may be present where performance and requirements cannot be established. In these latter cases experiments must be defined which can either (1) furnish the needed information by the simulation of space conditions on the earth, or, (2) define requirements by actual space flights. It is this latter catagory of experiments requiring space flight which is of interest here.

If animals are to be used in space flight experiments, the experimental design must be such that the data are applicable to man. The application of such animal data will always require extrapolation or interpolation. In general, the selection of the experimental animal and the experimental design must be based upon extensive ground experience in order that such data may be meaningful. New and novel experiments tried for the first time in space may lead to erroneous conclusions because of lack of experience. Until such time as extensive animal or human experiments can be performed in space, judgments and decisions, based upon the meager space experiments to date, must perforce be largely subjective.

6.3.3 Experimental Design

In order to illustrate the considerations in the design of an experiment for the study of prolonged weightlessness the following general remarks may be made. Three problems are of general interest in connection with zero g: (1) the effects of zero g per se, (2) adaptation to zero g, and (3) readaptation to gravity upon return to the earth.

The experimental conditions require that the zero g state shall be reached from one g via accelerations greater than one g. The condition of going from zero g back to one g also requires a transient phase of greater than one g acceleration.

In general, zero g may or may not have measurable effects. If experimental observations indicate that zero g has no biological effect, such a result may be because either (1) there is, in reality, no effect, or (2) the adaptation of the biological organism to zero g has been immediate with no noticeable time lag. For instance, although the human body may experience severe hemmorrhage, there may be no change in arterial blood pressure if conditions are arranged appropriately. This is because the body compensates for the loss of blood and maintains arterial blood pressure by regulatory mechanisms. A similar situation must be considered under zero g.

If exposure to zero g results in noticeable effects, these effects may be a function of g with no adaptation, or they may be a function of g and time because of the occurrence of natural adaptation. The results of exposure to zero g can only be ascertained by the study of measurable functions and such functions may display either a short or long time constant. One of the purposes of the experimental design should be to elucidate these time constants.

If natural adaptation does not occur, or if natural adaptation has a long time constant, adaptation of some kind may be required. The nature of the adaptation may only be apparent upon return to the earth. This is because it has been frequently observed that the time constants of readaptation may be noticeably different from those of adaptation. This effect is illustrated in Figure I-6-4, curve (c).

It is apparent that both physiological and performance measures are required in order to define tolerance to an environmental effect such as zero g. In the case of studies of carbon dioxide in submarines it has been demonstrated that certain carbon dioxide levels are associated with physiological effects but with no performance effects. Other levels are associated with performance effects but no measurably physiological effects. At high levels both physiological and performance effects are noted. Thus, several echelons of effects may be studied. These vary from such extremes at death down through such levels of tolerance as unconsciousness threshold to high stress. At the other end of the scale are effects associated with low stress on the biological

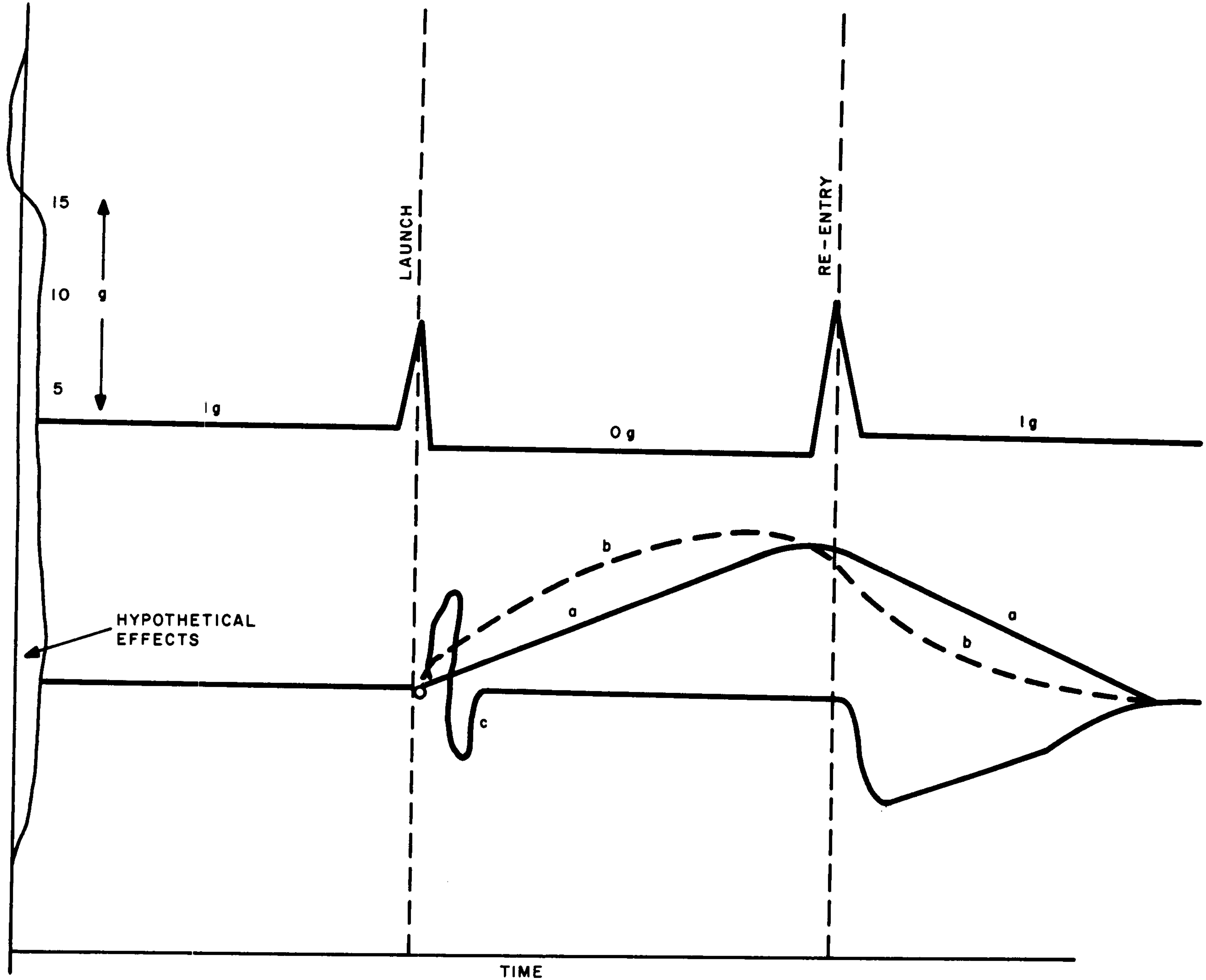


Figure I-6-4. Adaptation/readaptation effects subjects. Under these latter circumstances scientific evaluation is extremely difficult and depends upon the elucidation of mechanisms of the effects and their associated bodily responses.

If the primary objective in an experimental design is to be achieved, all environmental parameters must often be controlled with the exception of the one whose effect on a system is to be measured. For example, in order to separate the effects of weightlessness of long duration from those of ionizing radiation in space, orbits beneath the Van Allen belts must be chosen or extensive shielding provided. Ideally, if important interactions are likely to occur, the effects of ionizing radiation in space should be determined on the ground, if the types and energies of the particles in space could be accurately duplicated, thus eliminating weightlessness as a contributing factor to the experimental data. Although accelerators are not available that will accelerate particles to the energies required, data obtained on the ground are obtained under one g. In order to study radiation at one g in space a new variable, Coriolis forces (which can only be reduced to low levels by rotating vehicles at very long radii), would then enter the experimental design. One advantage in the utilization of space vehicles as research and test stations however, is that the interactions of a number of variables (acceleration, vibration, radiation, etc.) can be studied at one time. Thus, after various parameters have been studied singly, they may be combined in space for a study of possible additive effects. Current ground facilities will not permit such complex experiments.

6.3.4 Bioinstrumentation Program

The overall bioinstrumentation program for Project APOLLO may be divided into three phases: (1) initial operational assurance testing; (2) prolonged orbital flight; and (3) the lunar voyage. In the initial phase crew safety will require monitoring. In the orbital phase detailed biological studies will be conducted. The bioinstrumentation requirements for the lunar voyage may be negligible and the emphasis placed on the mission instrumentation.

For the biologist the orbital phase of Project APOLLO holds as much interest as the lunar flight itself because the orbital flight program will be the biological testing ground for the lunar flight. Careful consideration must be given during this period as to the means of verifying what are at present somewhat speculative biological parameters. A major feature of the orbital flight program will be the detailed investigation of the effects of prolonged weightlessness on man.

6.3.4.1 OPERATIONAL ASSURANCE AND MONITORING

In the initial flight test program there will be a requirement for monitoring the well-being of the crew. Monitoring may be defined as the accumulation and reduction of data for the purpose of taking action. Two types of action may be contemplated: (1) on the basis of the information gathered the flight may be aborted in the interest of crew safety; and (2), although crew safety may not be in jeopardy, the accumulated data may lead to redesign and retrofit of the life support system or other vehicular mechanism. In the first case it is assumed that the values of the biological parameters measured can be interpreted in terms of the overall state of the individual and implies that these values can be synthesized in terms of the physiological well-being of the crew. The second case implies that the crew can be used as an omnitransducer of the system and that the measurements can be interpreted in terms of the hardware functions which support life in the vehicle. Both of these suppositions must be examined closely in terms of the selection of parameters to be measured and the reduction and display of this information for crew and/or ground-based action.

The selection of biomedical parameters for Project Mercury has been based almost exclusively on two considerations, namely, the availability of transducers and techniques readily adaptable to the stringent requirements of actual space flight and an emphasis on the clinical aspects of monitoring. In general, many laboratory methods and devices in common use can not be qualified easily and utilized without extensive development and, even if developed, would markedly encumber the crew in the performance of their duties. The measures which have been selected for use initially in such programs as the X-15 and Mercury are the electrocardiogram (which primarily measures heart rate), the respiratory rate (with only a qualitative indication of respiratory depth) and either core or a skin temperature. Other measures under study include the measurement of blood pressure, the psychogalvanic reflex and the electroencephalogram. It is important to recognize the rather limited range of such biological variables in characterizing the well-being of the astronaut.

The problems of using the heart and respiratory rate for monitoring have recently been highlighted in the chimpanzee experiments of Project Mercury. Detailed studies at Holloman AFB were required to establish normal values and ranges for young chimpanzees, and expected variations were established by extensive simulation of the flight environment. While such studies are technically of interest no action can be taken during the ballistic flight and it is doubtful whether the data would be of value in the diagnosis of equipment malfunction. The study of the psychomotor performance of the animal would appear to be of more general value. Performance of the animal reflects its overall response to the environment if the tasks are selected with care. Such tasks must be sufficiently sensitive to reflect environmental changes, however. It is possible to design the task regime in such a way that no decrement in performance occurs until unconsciousness supervenes.

Environmental stress may result in physiological changes alone, performance changes alone, or combined effects. Without measures of performance biological measures of the response to stress may be interpreted primarily in terms of viability. Because of the inadequacies of current bioinstrumentation, crew safety must depend strongly on data accumulated from environmental sensors and upon voice communication. Although the environment may be normal, internal stresses such as fear and anxiety may occur. The likelihood of such stresses may be lessened by crew selection and training. Voice communication, however, provides an important link. The verbal output of the crew represents an extensive data reduction from the variety of natural sensors available to man and, taken together with data on the environment, the crew performance and a few simple biological measures, represents adequate data for preserving crew safety. The chief failing of the voice link occurs in states which cloud the sensorium. Under these circumstances inconsistencies in verbal reports and biological signals may provide an insight into the true conditions.

The requirements for on-board data reduction has been discussed by McLennan. Using the GSR he examined the possibilities for conserving bandwidth and power. If only a few items are to be telemetered and if adequate power is available, data reduction and display may more easily be conducted at ground-based installations. The design and manning of such ground-based monitoring stations has been the subject of considerable study in the Mercury program. In order to achieve the objective of continuous coverage of the Mercury astronaut in his first orbital flight, a network of monitoring stations has been established and will be manned by flight surgeons. Although data will be displayed and voice communication possible at each station, final decisions will be made at a central command station where all information will be available. Decision to abort must be made with respect to the landing site and availability of search and recovery facilities. Even in the event of an onboard catastrophe these factors cannot be ignored if recovery is to be effected.

In the Mercury ballistic flights and in the X-15 flights no abort alternative is available following launch. In X-15 flights to date pulse rates have varied according to the stress of the various flight phases (drop-off, ignition, burnout, re-entry G and landing had high pulse rates). The monitoring of space suit pressures is of considerable significance. For proper function, helmet pressure must always exceed suit pressure in the X-15 configuration. Alterations or reversal of this pressure gradient would call for the use of the emergency oxygen supply.

For the lunar voyage, little or no requirement for crew monitoring by bioinstrumentation is conceived. The extensive flight testing of the vehicle in orbit which will precede the lunar trip should alleviate this necessity. Emphasis will be placed on the voice link and probably TV coverage. Any measurements which may be required can be made on ground command by one crew member on another. It seems probable that ground-based as well as on-board monitoring of the gaseous and physical environment may be desirable in order that the ground may assist in the maintenance and repair of the environmental control or other system if required.

6.3.4.2 ORBITAL PHASE

The major new bioinstrumentation requirements for Project APOLLO arise because of the extensive biological studies which will be needed during the orbital phase of the flight program. Initial flights will require extensive vehicle instrumentation in order to prove out the vehicle and its subsystems. Instrumentation for the lunar flight will be concerned with observations and measurements of the moon. During the orbital phase the APOLLO vehicle will become, in part, a biological laboratory. The major biological study to be undertaken will be that of the effects of prolonged weightlessness on man. Other experiments will be concerned with the biological effects of radiation and biological rhythms.

6.3.4.2.1 Prolonged Weightlessness

In order to study the effects of prolonged weightlessness on man biochemical techniques must be developed for possible use in orbit. In general, successful conduct of the experiments required more than one member of the crew. Techniques for the conduct of experiments must make use of small quantities of blood, plasma or urine and the methods must be adaptable to use under zero g. Such techniques can make use of blood samples from the finger or ear lobe drawn up in capillary tubes. A number of micromethods have been developed for studies of glomerular functions which may be applicable (Richards) and radioactive measurements may lend themselves to zero g application. Such biochemical measurements will be supplemented with clinical measurements made by one crew member on another.

Significant areas for study include the general metabolic balance, the problem of muscle atrophy, bone demineralization, reflex effects of blood shifts and blood volume changes. An interesting basic area for consideration is the study of otolithic and semi-circular canal functions and their interactions in the absence of gravitational vector.

6.3.4.2.2 General metabolic balance.

Confinement studies have generally demonstrated a marked reduotion in metabolic rate and, because of the absence of gravity, there is reason to believe that metabolic demands will decrease as the result of a decrease in proprioceptive reflex activity. Metabolic data are needed in order to refine the requirements for the environmental control system, particularly storage requirements and purification capacities.

Metabolic data may be acquired by measurements of oxygen consumption from the vehicle stores. Such data will reflect only the average activity of the crew and will not provide detail on metabolic rates during sleep or those characteristic of task performance. In addition, oxygen consumption rates must be corrected for gas leakage. Leakage rates can be calculated from measurements of a diluent gas concentration such as nitrogen.

Dietary intake aboard the vehicle will be well controlled. Individual measurements of oxygen-consumption could be conducted using a light-weight wedge type spirometer in the mission module. Individual dietary intakes and oxygen consumptions for various ranges of activities combined with average data for the entire crew should give a good picture of the metabolic requirements of weightless space flight.

6.3.4.2.3 Muscle Atrophy

Although general metabolic data may be helpful in evaluating the status of man's muscle, more detailed information should be acquired. A number of simplified methods are available:

1. Measurement of muscle girth (calf or biceps) by one crew member on another requires only a tape measure.
2. Muscle strength could be evaluated with an ergometer consisting of a spring loaded device rather than a pulley and weight. Grip strength is easily evaluated by a small light weight device.
3. Electromyographic studies should be studied for possible application. A similar electrical amplifying and recording set-up would be required for ECG, EEG, EMG and nystagmographic studies and could be used interchangeably.
4. Chronaxie measurements would require a simple stimulator and should be considered
5. Perhaps the most difficult but valuable measurements would be an estimate of urinary nitrogen excretion and this is considered below.

6.3.4.2.4 Bone demineralization

Two means for following the process of demineralization are available: before and after x-ray studies and measurements of urinary calcium excretion.

6.3.4.2.5 Blood Volume Changes

In order to follow blood volume changes urinary output must be measured. Daily urinary volume measurements will also be required to follow nitrogen and calcium losses if they should occur. Urinary sodium loss is also a desirable measurement.

Direct measurement of blood volume presents many difficulties. Whether the injection of materials into the blood stream for dilution studies is warranted is questionable. Decision on this matter should await the interpretation of other data.

6.3.4.2.6 Vestibular Function

Consideration should be given to the design of experimental studies of semicircular canal and otolithic function. In space there is an opportunity to study these function independent of the gravitational vector. In addition, the effects of vision and proprioception can be isolated to some degree. Available experimental data which relates electro-myographic and nystagmographic recordings as well as illusionary phenomena to these functions should be carefully scrutinized in order to establish a workable experimental design in orbit in terms of weight and space requirements.

Many of the experiments suggested involve the collection and analysis of fluid samples. It is proposed that blood and urine samples be collected in labeled capillary tubes. Such tubes could be conveniently stored in the frozen state for subsequent analysis. Consideration, however, should be given to analysis in orbit. Problems arise with respect to crew training and methodology. It would seem to be possible to teach crew members to collect blood samples from the ear lobe, to measure the daily urinary volume and to collect urine samples. Whether the analysis is conducted in orbit or after re-entry, an intensive study and proof of suitable micro-methods is in order.

6.3.5 Biological Rhythms on Space

Since the time of DeMairan, over 200 years ago, there has been scattered interest in the biological rhythms exhibited by plants and animals. Most of the interest in recent years has been focused on circadian (about one day) rhythms. There are numerous examples of these types of rhythms, body temperature, eosinophil count, mitotic rates, etc. These short-term rhythms are, in turn, superimposed upon rhythms of longer periods such as monthly (menstrual cycle) rhythms and yearly (bird migration) rhythms.

All of these rhythms are correlated with the rhythmicality of the earth's environment, which is at least in part affected by the earth's movement relative to the sun and moon. The earth's rotation relative to the sun gives a 24 hour day; rotation relative to the Moon and its revolution about the earth gives a lunar day of 24 hours and 50 minutes;

the Moon's arrival every 29.5 days at the same relative position between the earth and sun marks the so-called synodical month; the earth revolves about the sun every 365 days, 5 hours and 48 minutes. The daily and annual rhythms related to the sun are associated with changes in light and temperature. The 24.8 hour lunar day and the 29.5 day synodical month are associated with the moon-dominated ocean tides and with changes in nighttime illumination. All four types of rhythms include changes in forces such as gravity, barometric pressure, high energy radiation, and magnetic and electrical fields.

Obviously a correlation, even if a strong one, does not definitize a cause and effect relationship. Biologists are reluctant to credit a phenomenon for which there is no plausible explanation. In this case what is the receptor for magnetic fields? The literature contains many reports of the effects of light, barometric pressure, and of ionizing radiation; these may readily be related to biological receptors.

When one encounters rhythmical activity in some system it is necessary that some clock mechanism be present. This mechanism may be intrinsic or extrinsic. In the case of biological clocks the differentiation of the clock mechanism has not been made. Most researchers in this area are in favor of the intrinsic clock hypothesis. In fact various models of oscillators have been proposed. One may even devise a model to account for the strange temperature and drug insensitivity of biological rhythms. On the other hand, Brown at the University of Northwestern is of the opinion that the clock is essentially external or a combination of an external plus an internal clock.

If we suppose that the clock is intrinsic and that all stimuli are excluded, then it might be expected to "free-run". On the other hand, if the clock is external, the internal mechanism may be likened to a one-shot multivibrator; i.e., if no synchronization input is present, there is no output. Brown has shown that animals placed into an environment, where light, temperature, and pressure are constant, still exhibit rhythms and that these rhythms may be altered by moving these animals from one part of the earth to another. What happens when an organism is not only moved from one part of the earth to another, but when it is removed from the surface of the earth? The animal is now in another orientation with respect to the earth, sun, and Moon and one may expect alterations or cessation of various rhythms. However, if the clock is internal, as the majority suppose, then no or slight alterations are to be expected. If the clock is external, then radical changes may occur.

What does this mean to the astronaut? Physiologists and psychologists alike have performed considerable research on work-rest cycles and have fairly well established cycles of various periods that may be utilized in circumstances where a small crew must operate continuously for extended periods. Naturally, all of this work has been performed on earth, or at least, very close to it. Are these findings valid in space vehicles? Other parameters such as susceptibility to infection and stress tolerance may also be affected.

Although the foregoing discussion has been based on very little evidence, in view of the findings of Brown at Northwestern, Beischer at Pensacola with shielded rooms, and some unconfirmed Russian reports of slight metabolic changes in space vehicles it would appear that some attention should be focused on this problem. Adequately instrumented animal subjects in space vehicles should prove a useful tool in solution of the biological clock problem.

6.3.6 EQUIPMENT CONSIDERATIONS

As has been pointed out in a previous section, the major application of biomedical instrumentation will be required in the earth-orbital development phase of the APOLLO program. Since that phase is relatively soon, only minor instrumentation development can be allowed. Fortunately the basic instrumentation for Mercury will satisfy most of the instrumentation tasks previously discussed.

More detailed discussion of bioinstrumentation equipment is found in Volume VII.

7.0 Radiation

7.1 INTRODUCTION

The APOLLO radiation study has been directed at the solution of the principal problems arising from the consideration of Van Allen belt and solar flare radiations. The primary objective of the study was to determine the biological dose to which crew members will be exposed and to develop methods of limiting the dose to a safe, permissible value of 25 REM as specified in the study guidelines. Toward this end, analyses were undertaken to evaluate the integrated dose encountered in passage through the Van Allen belts on trajectories typical of the APOLLO mission. The consideration of solar flares led to the evaluation of vehicle design from the point of view of radiation shielding, and to an analysis of the solar flare hazard on a probability basis. In addition to the study of the trajectories and vehicle design, consideration was given to physiological techniques for ameliorating the radiation hazard.

7.2 VAN ALLEN-INTEGRATED DOSE STUDY

7.2.1 Introduction

In this phase of the radiation study, three classes of trajectories were investigated:

1. Lunar Ascent
2. Emergency Return
3. Earth Orbit

A number of examples of each type were analyzed; the integrated dose and dose as a function of time were determined for each trajectory. The results of these investigations and a description of the program which was employed follow.

7.2.2 Integrated Dosage Program

The digital computer program written for this analysis accepts initial position and velocity information and generates a point by point two-body solution of the trajectory equation. A model of the radiation belts is stored in memory in matrix form giving dose in roentgens/hr as a function of geomagnetic latitude and distance from the mid-point of the earth's magnetic dipole. The program records the dose rate at each point and integrates the maximum value of dose rate in each time interval, thereby yielding an upper estimate of the integrated dose.

7.2.3 Assumptions

The model of the Van Allen belts is based on the data from the Minnesota Experiment in Explorer VI, passes 1-14 August 7-14, 1959. The data is extrapolated by assuming symmetry with respect to the axis of the geomagnetic dipole and with respect to the geomagnetic equator. This extrapolation is based on the assumption that the dominant factor in the particle-trapping phenomenon is the magnitude and direction of the earth's magnetic field, and that the corresponding symmetries will obtain. Interpolation in intensity is taken in accord with counting rate data obtained by instruments in the Pioneer III and Explorer IV satellites. The ionization data from the Explorer VI corresponds to a measurement taken behind approximately 1 gm/cm^2 of shielding, and as such provide a conservatively high estimate of the level encountered by a man in a space vehicle. Particularly, the dose levels in the inner Van Allen belt would be considerably reduced by the nominal amount of shielding 2-6 gms/cm^2 provided by the vehicle structure (Schaefer). To simulate cosmic ray background radiation a value of .0006 R/hr was employed. Spatial variation of this radiation was considered to be of negligible effect and the constant value representing the peak cosmic ray dose rate of .0006 R/hr was taken at all points outside the radiation belts.

For purposes of evaluating the biological dose in REM from the ionization data, a ratio of one REM per roentgen will be approximately correct. The principal error incurred in this approximation arises when considering the dose contribution from the inner Van Allen belt protons. In this case an appreciable number of the incident protons are in the energy range in which the RBE is greater than one. However, as indicated above the conservative estimate of the effective shielding inherent in the data compensates for this error. In the outer belt the assumption is subject to the variation in the energy absorption of tissue (nominally 93 ergs/gm per roentgen) over the range of photon energies 10 KEV - 800 KEV (Hine and Brownell). This variation is small and well within the uncertainties inherent in the experimental data.

The data obtained in the investigation of integrated dose are presented in the form of two graphs for each trajectory. The first in each case is a trace of the trajectory superimposed upon a map of the Van Allen belt equal dose rate contours. The coordinates are geomagnetic latitude and distance from the center of the earth.

7.2.4 Results

7.2.4.1 LUNAR ASCENT TRAJECTORIES

Four lunar ascent trajectories were run using initial conditions representative of Saturn burn-out data on launch from Cape Canaveral. In each case, the trajectories were run until the position of the vehicle was well outside the limits of the Van Allen belts. The total transit time for passage through the belts was approximately 2.45 hrs for all four cases. Table I-7-I gives the input conditions and total dose after 5 hrs. Definition of symbols in Table I-7-I.

V = Velocity ft/sec

γ = path angle

r = altitude st. miles

β = latitude

ψ = azimuth

λ = longitude

R_T = integrated radiation dose in roentgens

TABLE I-7-I. LUNAR ASCENT TRAJECTORIES

Case	V	γ	r	β	ψ	λ	R_T
A	36240	0	4019	9.045	127.0	-48.3	1.01
B	36019	0	4019	9.298	125.1	-45.6	1.09
C	36141	0	4019	9.045	127.0	-48.3	1.06
D	36341	0	4019	9.045	127.0	-45.6	1.01

In all cases the dose was approximately 1 roentgen which is satisfactorily low. A small increase in total dose with decreasing initial velocity is observable. Due to the similarity of the four examples only one is plotted in Figure I-7-1. It may be concluded that for nominal outbound trajectories, no modification of flight path is necessitated by consideration of the Van Allen belt radiation.

7.2.4.2 EMERGENCY RETURN TRAJECTORIES

The question of the dose encountered on an emergency return from an outbound trajectory was examined by simulating returns to earth from three different points along the ascent trajectory (Case B above). The normal Case B ascent was run until the desired altitude was attained. At that point a change in velocity and path angle was introduced to bring the vehicle back to earth. The change in the velocity vector was precalculated to produce acceptable re-entry conditions at an altitude of 400,000 ft. Return from an altitude of 3,998 statute miles above the surface of the earth resulted in an unacceptably high dose of 22 roentgens. Two alternate velocity vector changes were simulated in an effort to determine if a return from that altitude could be achieved with a lower total dose. The desired result was achieved as indicated in Table I-7-II. The results obtained from the emergency return simulation are shown in Figures I-7-2 through I-7-6.

Definition of symbols in Table I-7-II:

h_a = altitude above earth at initiation of return
V = resultant velocity
γ = resultant path angle
T_a = time to reach h_a

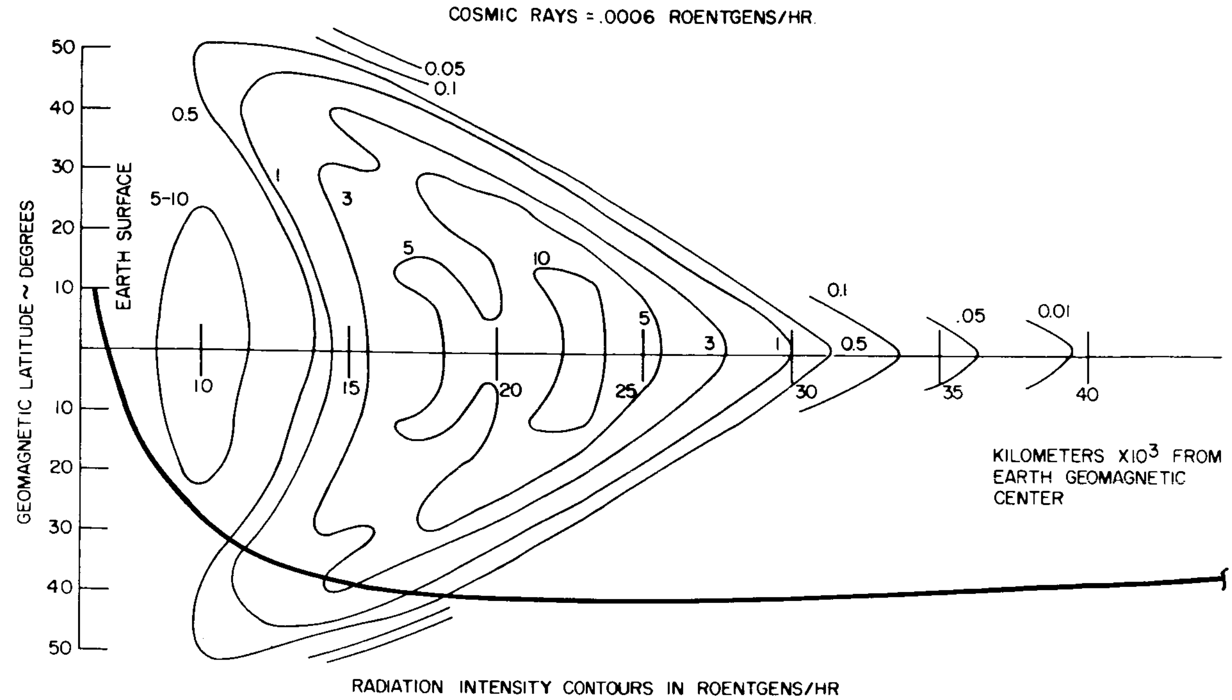


Figure I-7-1a. Lunar ascent trajectory - Case B

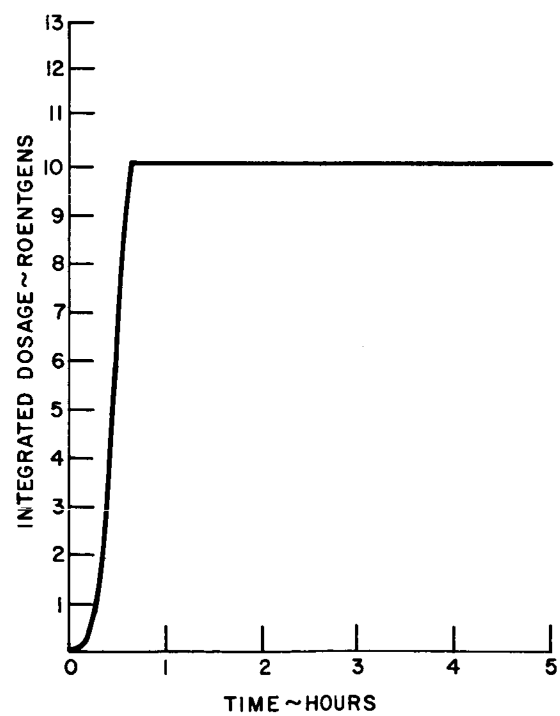


Figure I-7-1b. Integrated radiation dosage vs time lunar ascent - Case B

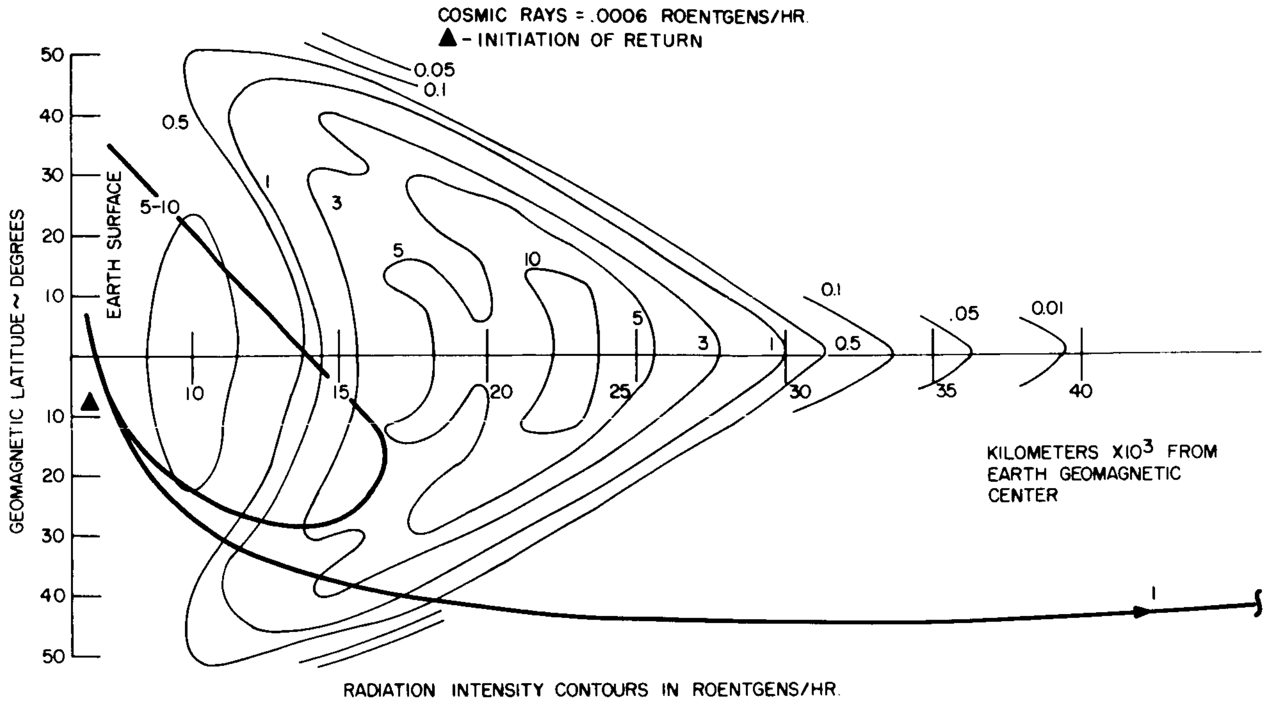


Figure I-7-2a. Emergency return trajectory - Case I

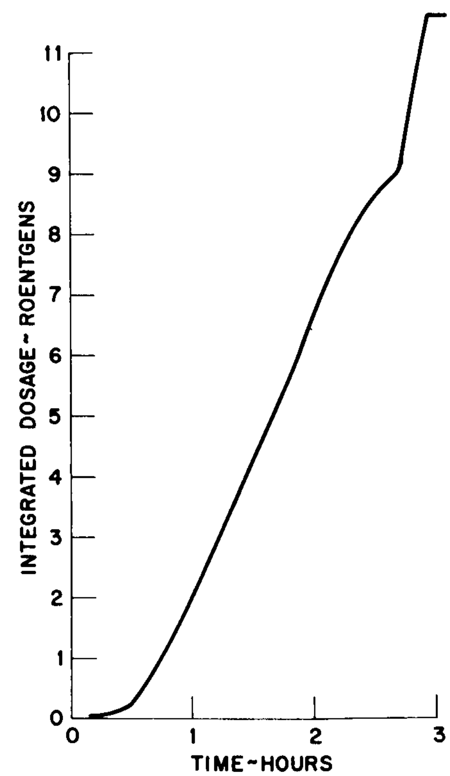


Figure I-7-2b. Integrated radiation dosage vs time emergency return - Case I

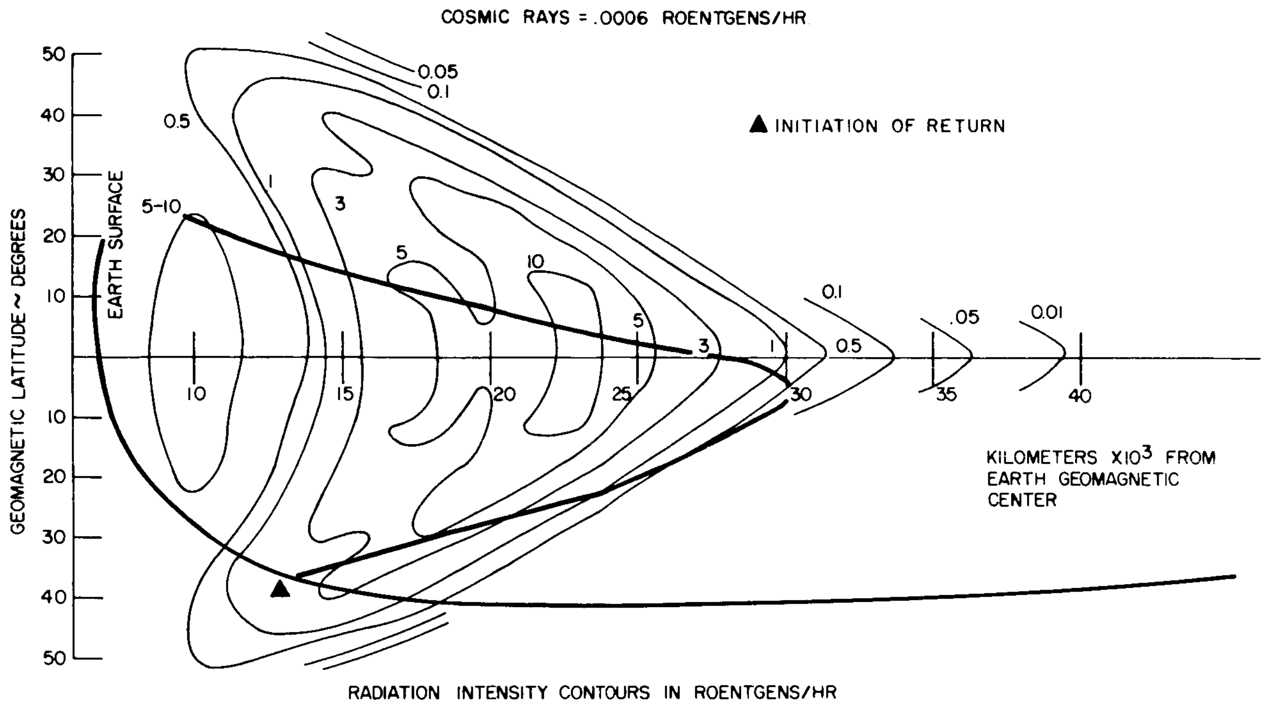


Figure I-7-3a. Emergency return trajectory - Case II$_A$

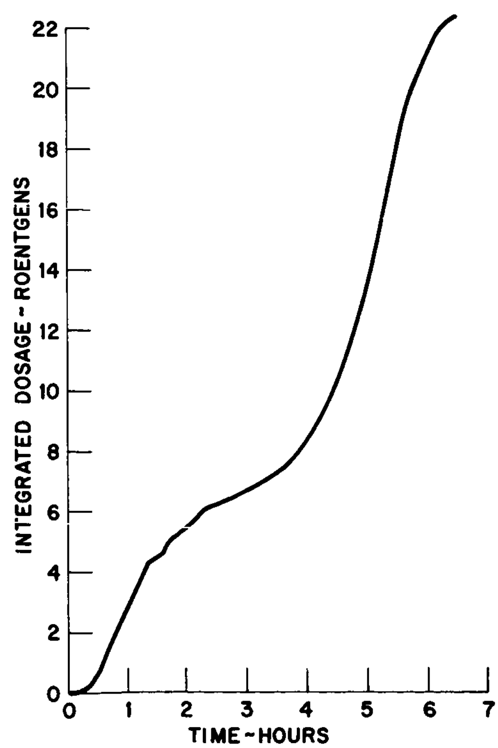


Figure I-7-3b. Integrated radiation dosage vs time emergency return - Case II$_A$

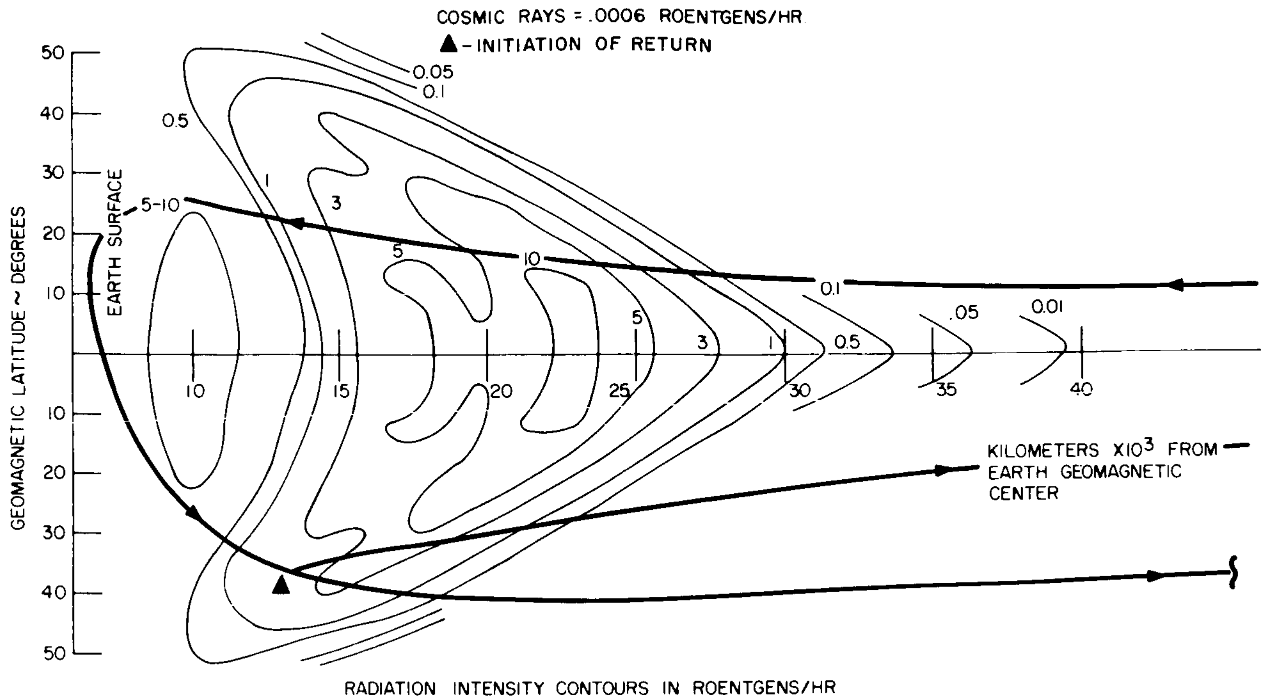


Figure I-7-4a. Emergency return trajectory - Case II_B

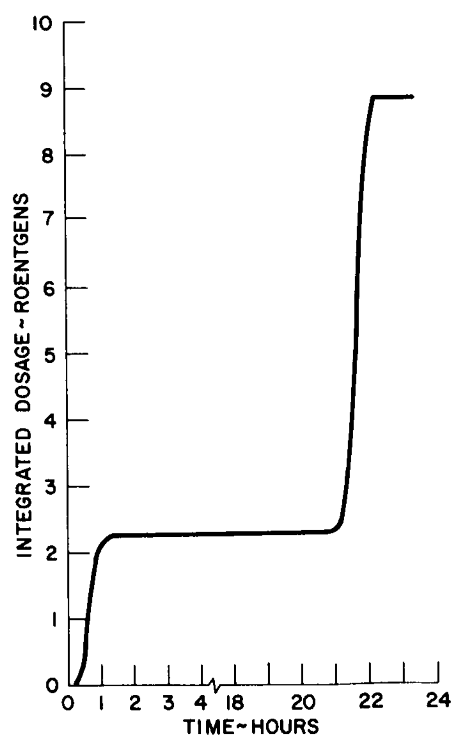


Figure I-7-4b. Integrated radiation dosage vs time emergency return - Case II_B

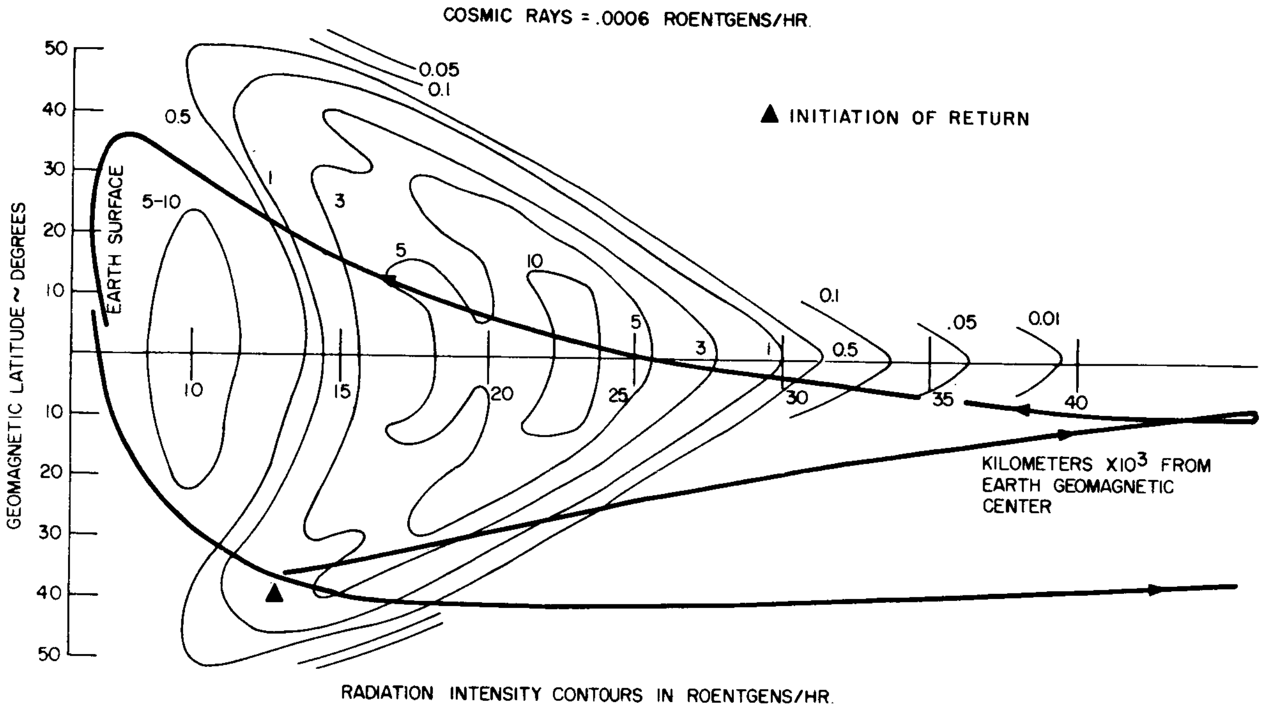


Figure I-7-5a. Emergency return trajectory - Case II_C

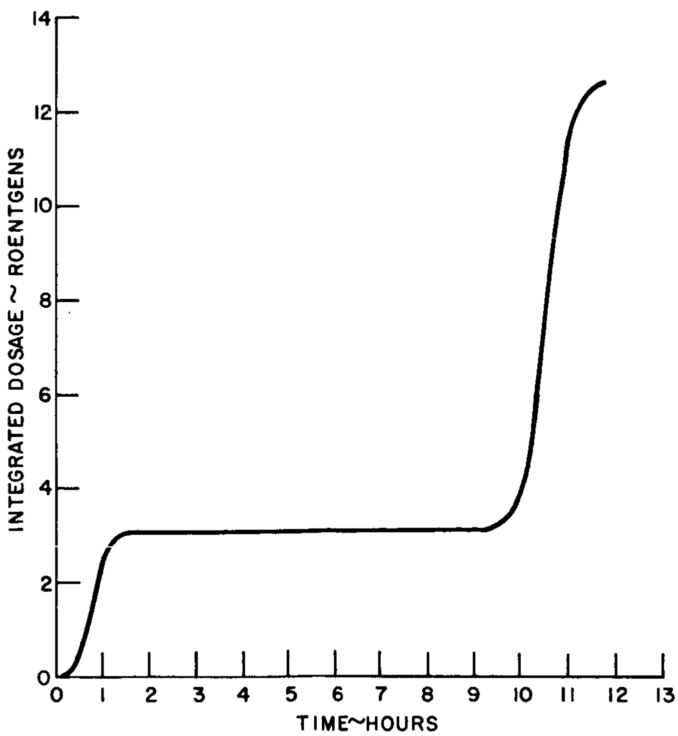


Figure I-7-5b. Integrated radiation dosage vs time emergency return - Case II_C

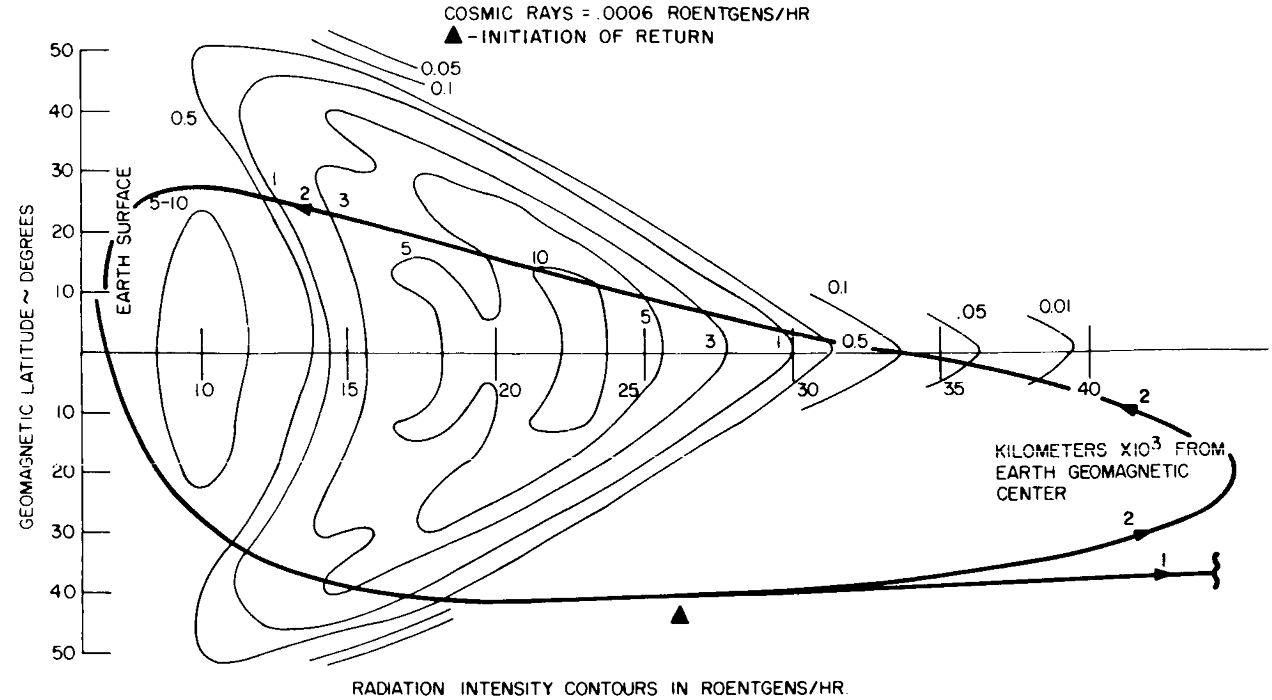


Figure I-7-6a. Emergency return trajectory - Case III

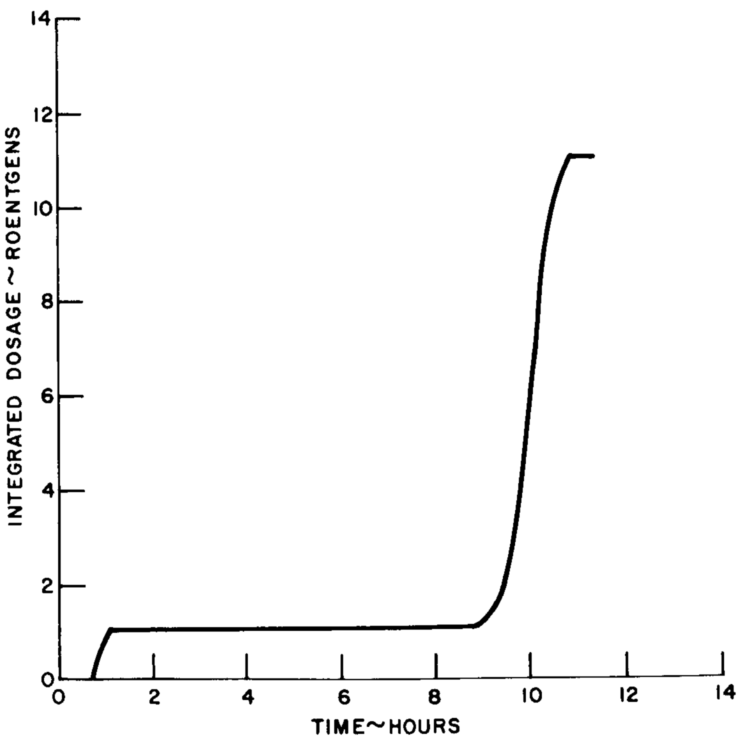


Figure I-7-6b. Integrated radiation dosage vs time emergency return - Case III

T_e = time to return to earth

R_T = integrated radiation dose in roentgens

TABLE I-7-II. EMERGENCY RETURN TRAJECTORIES

Case	h_a	V	γ	T_a	T_e	R_T Approach	R_T Return
I	345.5	29219	12.0	0.10	3.20	0.00	11.63
II_A	3998.2	20565	36.4	0.45	6.35	0.33	21.82
II_B	3998.2	23856	42.3	0.45	23.5	0.33	8.22
II_C	3998.2	22167	39.6	0.45	11.4	0.33	12.00
III	11641.1	12953	47.6	1.15	9.95	1.09	10.09

Referring to Table I-7-II it may be observed that an acceptably low dose is encountered in return from the three altitudes investigated. Case II_A resulted in a total dose of 22.2 roentgens. The alternate velocity changes represented by cases II_B and II_C gave a 50 percent decrease in total dose. The trajectory traces for the three returns from h_a = 3,998.2 st. miles permit a qualitative explanation of the result. In case II_A the vehicle path traverses the center of both Van Allen belts and does so at low geomagnetic latitudes, thereby resulting in what is probably an upper limit to the dose which might be encountered in an emergency return. The trace for Case II_B indicates that the trajectory does not pass through the highest intensity portions of either belt, and in fact avoids the inner belt entirely. For Case II_C the dose is increased due to traversal of the center of the outer belt at the geomagnetic equator. It may be concluded that for any given altitude a safe emergency return can be effected. The selection of velocity increment and resultant path angle, however, will affect the total path radiation dose and must be chosen subject to that constraint.

7.2.4.3 EARTH ORBITS

In order to establish the radiation hazard encountered in an earth orbiting trajectory a number of orbits were simulated. Cases Ø2 and Ø4 represent orbits generated with the same velocity, altitude and path angle but at different positions. The family of cases

1 through 5 represent orbits generated by insertion at the same position with the velocity varied to produce apogee altitudes from 20,000 to 60,000 n mi.

The resultant traces and dosage curves appear in Figures I-7-7 through I-7-13. The input conditions for the Cases appears in Table I-7-III. The data obtained are summarized in Table I-7-IV.

TABLE I-7-III EARTH ORBITS - INPUT CONDITIONS

Case	V_o	ho n mi.	β_o	λ_o	ψ_o
∅2	34,000	120	16.0	-47.0	118.1
∅4	34,000	120	9	-52.0	129.6
1	33,440	150	↓	↓	↓
2	34,125	↓			
3	34,512				
4	34,762				
5	34,936				

TABLE I-7-IV EARTH ORBIT DOSE

Case	Apogee Alt.	Period - hrs	Dose R-1 Orbit
∅2	54×10^3Km	14.2	10.50
∅4	54×10^3	14.2	2.10
1	45×10^3	11.5	5.05
2	64×10^3	19.9	1.00
3	84×10^3	27.8	7.15
4	105×10^3	37.0	2.43
5	126×10^3	47.4	2.40

For definition of symbols see Tables I-7-III and I-7-IV.

Comparison of the results of the ∅2 and ∅4 trajectories reveals that the change in position of the insertion point has a marked effect on the total dose reducing it from 10.5 to 2.1 roentgens. The difference arises from the facts that on the outbound portion of the

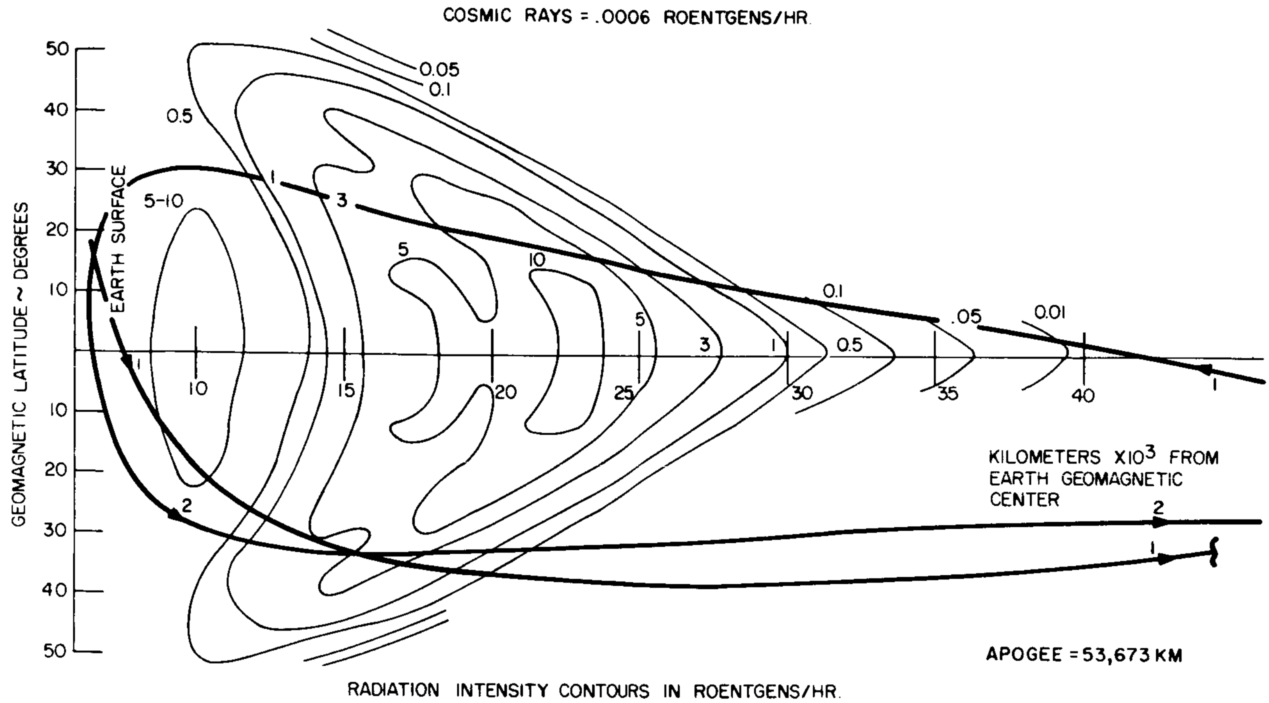


Figure I-7-7a. Earth orbit - Case Ø2

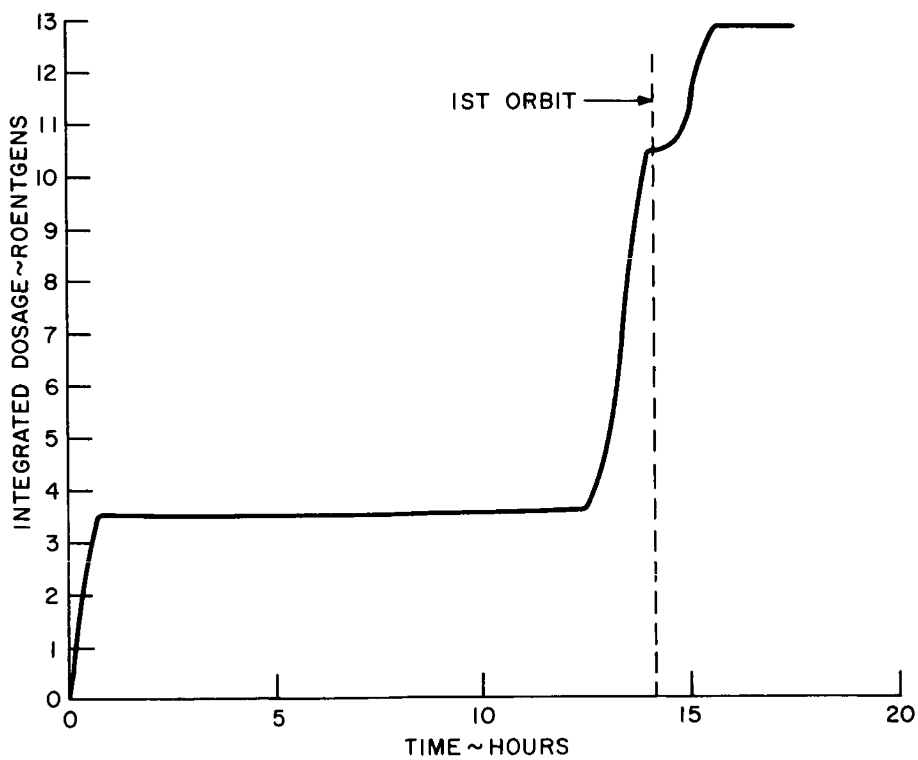


Figure I-7-7b. Earth orbit - Case Ø2 - Integrated radiation dosage vs time

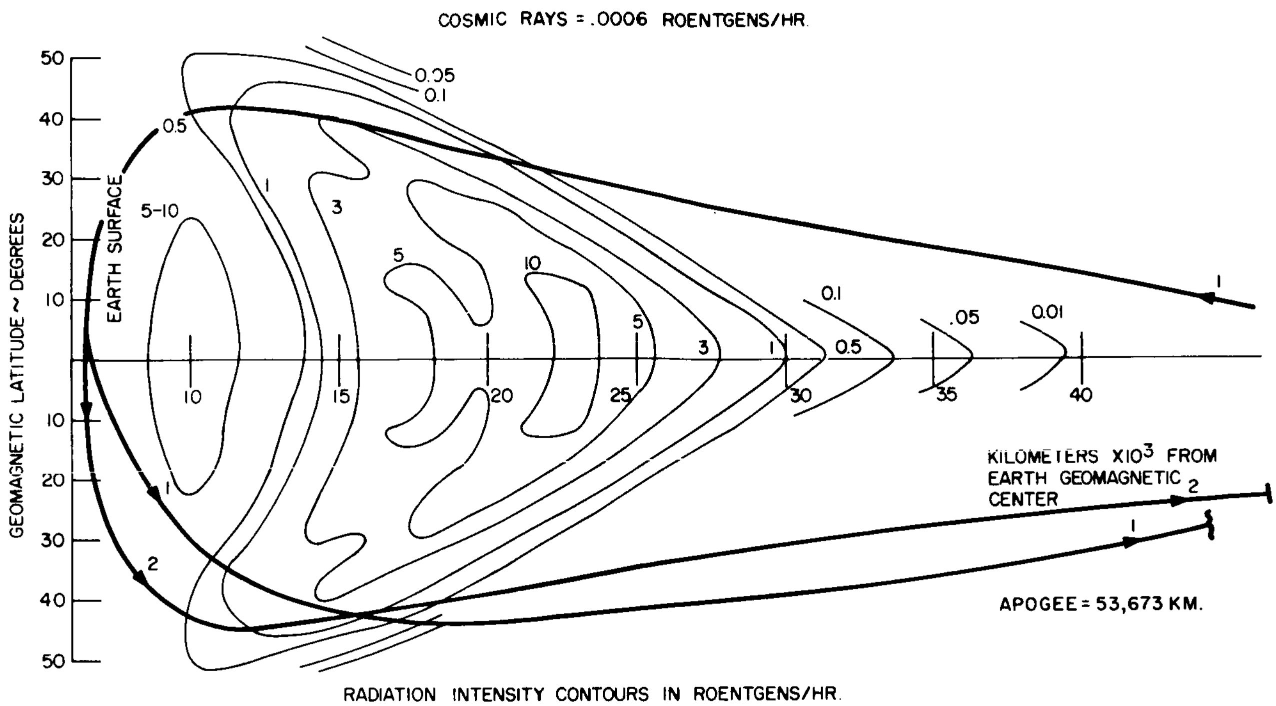


Figure I-7-8a. Earth orbit - Case Ø4

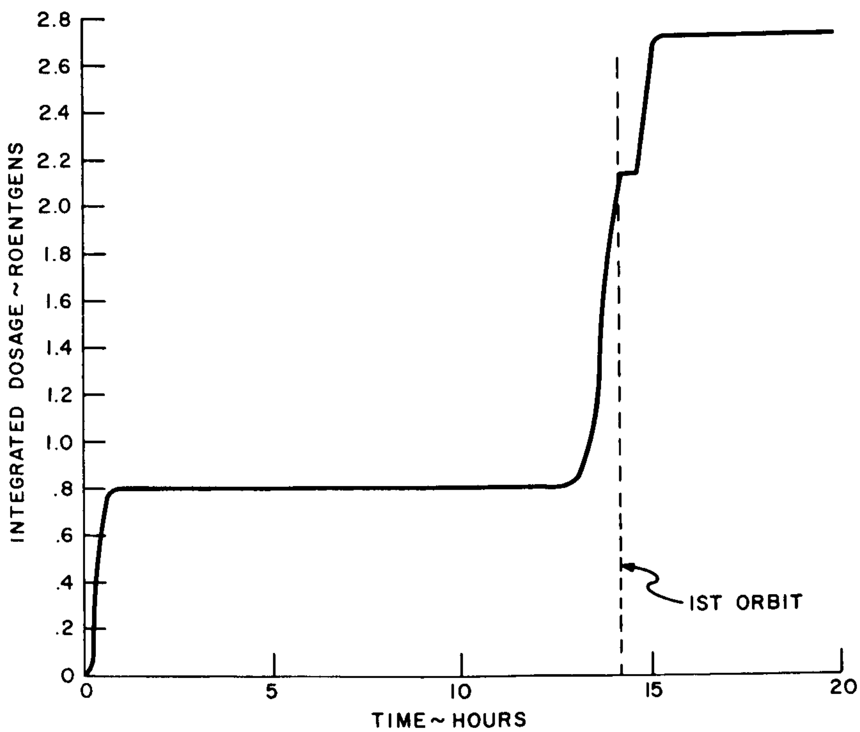


Figure I-7-8b. Earth orbit - Case Ø4 - Integrated radiation dosage vs time

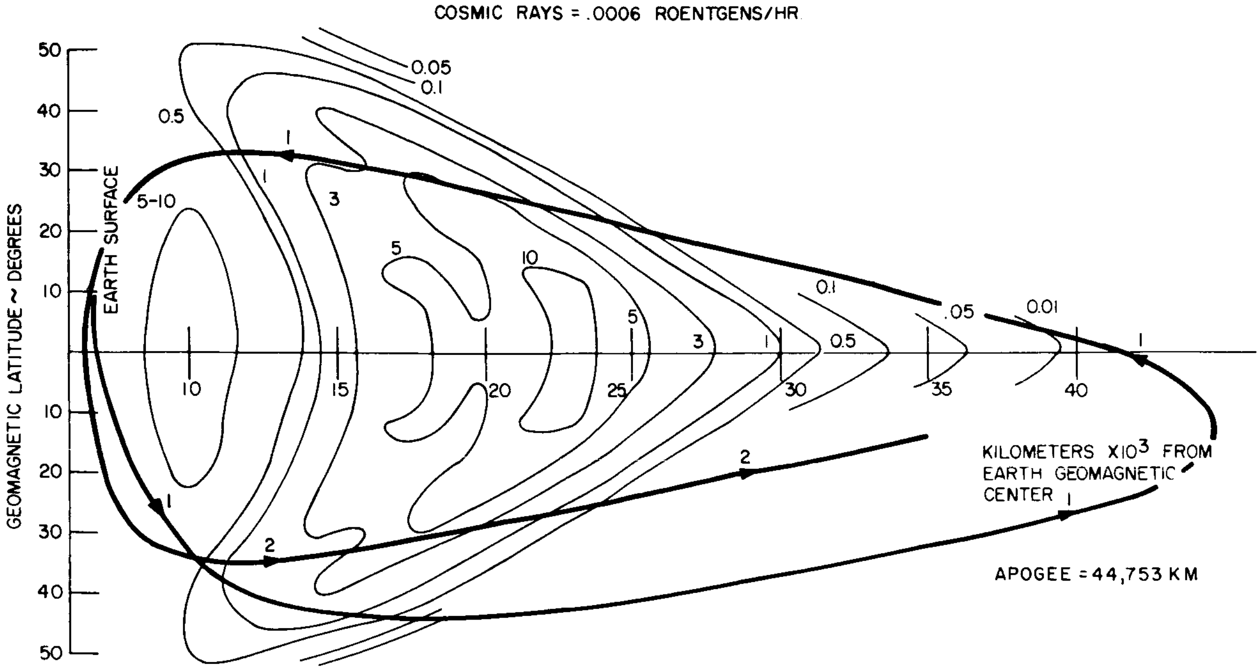


Figure I-7-9a. Earth orbit - Case 1

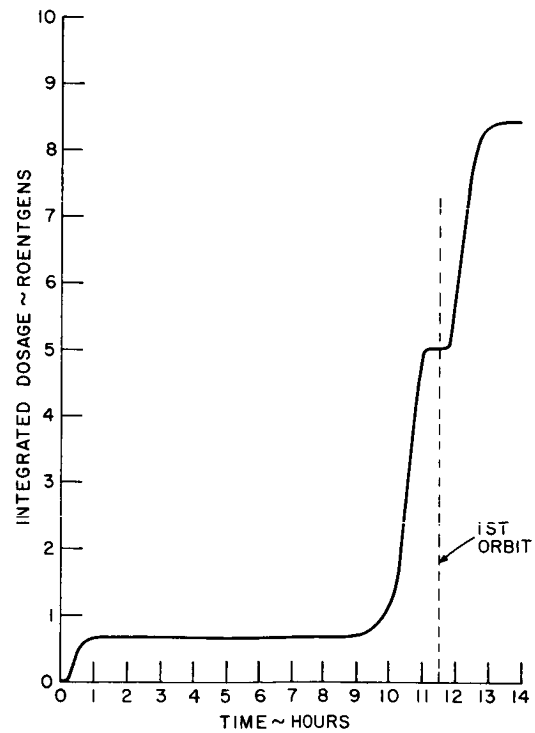


Figure I-7-9b. Earth orbit - Case 1 - Integrated radiation dosage vs time

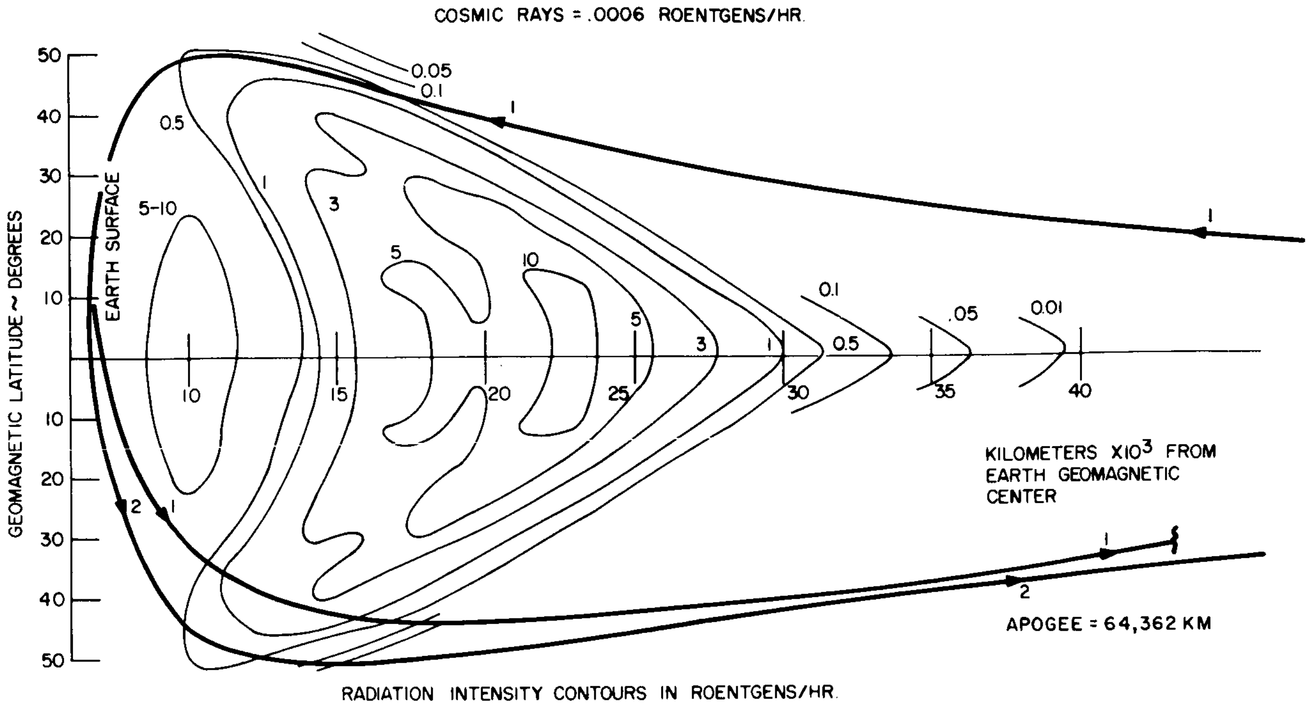


Figure I-7-10a. Earth orbit - Case 2

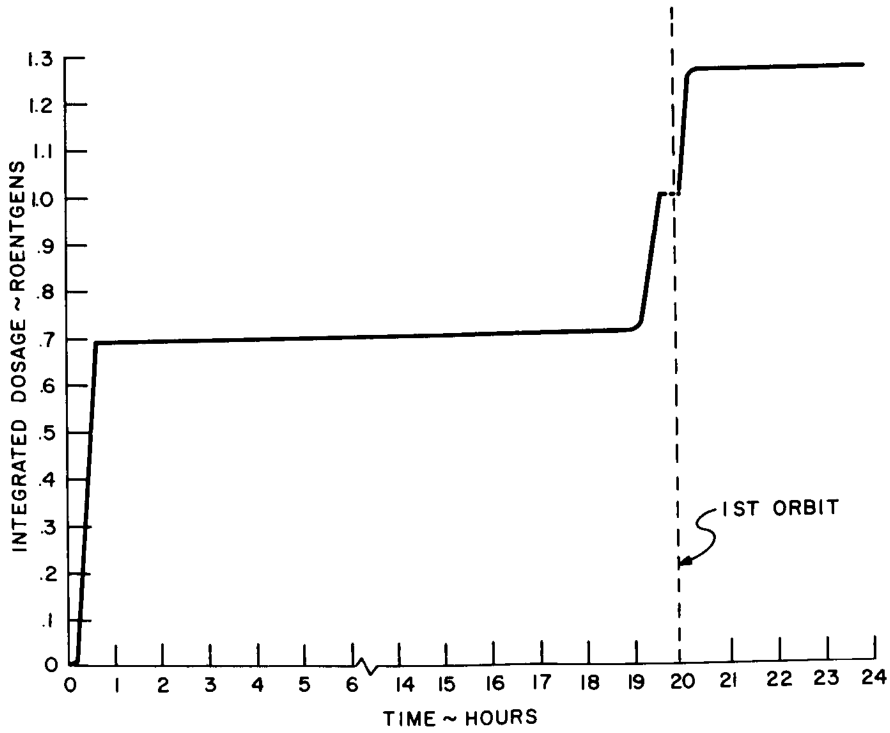


Figure I-7-10b. Earth orbit - Case 2 - Integrated radiation dosage vs time

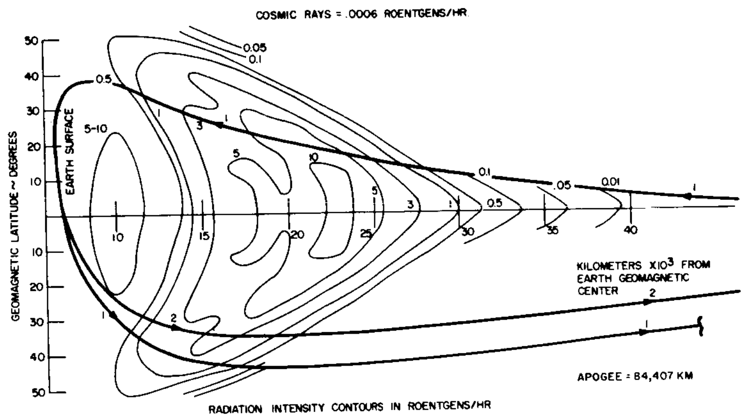


Figure I-7-11a. Earth orbit - Case 3

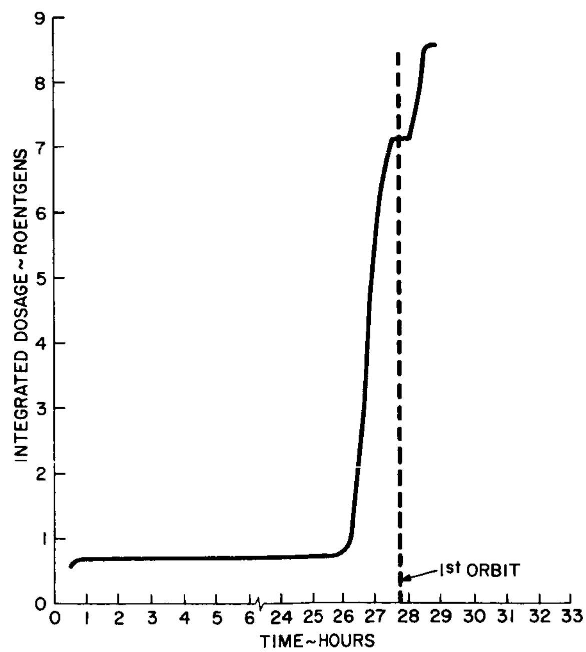


Figure I-7-11b. Earth orbit - Case 3 - Integrated radiation dosage vs time

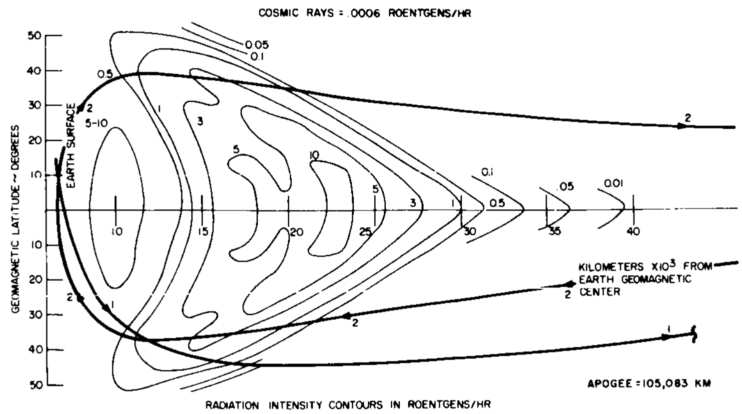


Figure I-7-12a. Earth orbit - Case 4

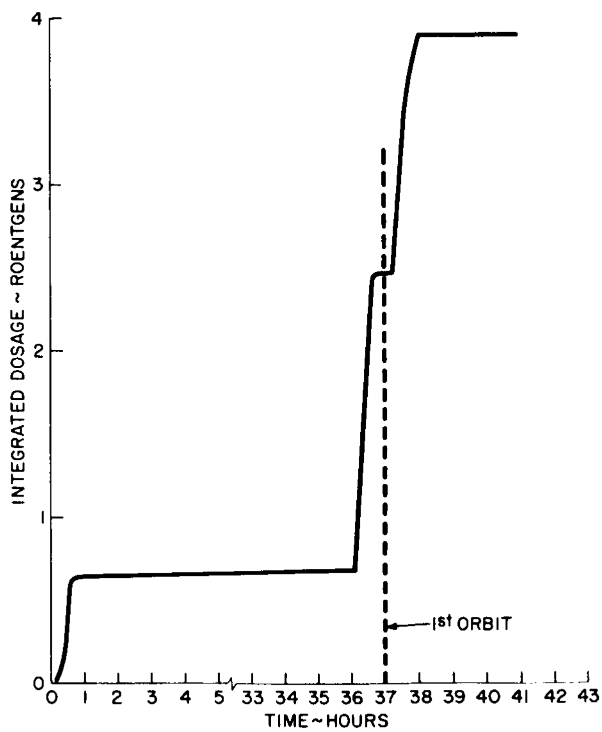


Figure I-7-12b. Earth orbit - Case 4 - Integrated radiation dosage vs time

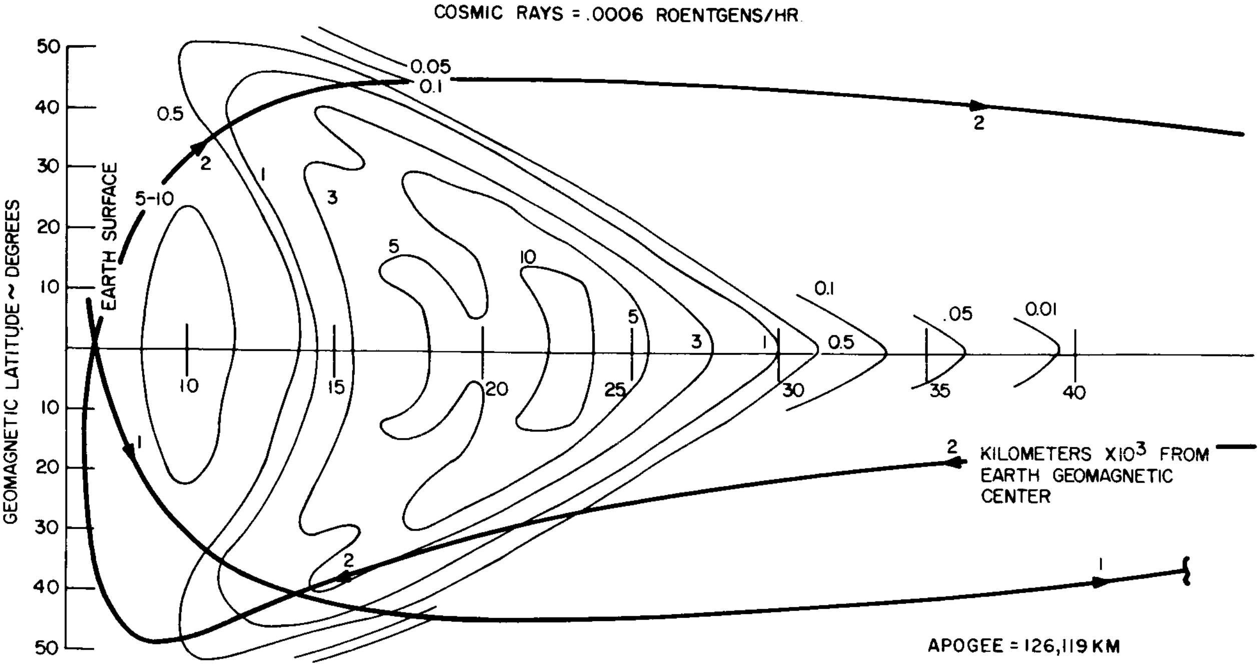


Figure I-7-13a. Earth orbit - Case 5

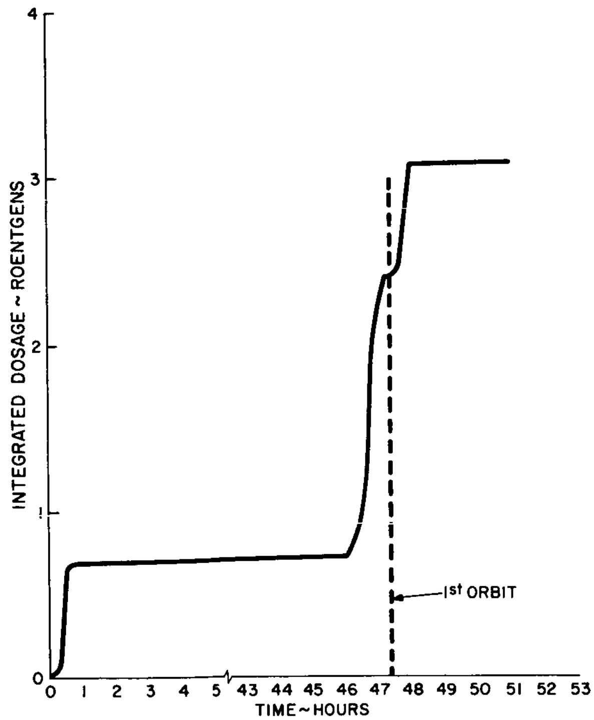


Figure I-7-13b. Earth orbit - Case 5 - Integrated radiation dosage vs time

orbit in the ∅2 case the trajectory passes through the center of the inner belt while in the ∅4 case the inner belt is not traversed. In addition, the return leg of the ∅2 orbit passes through the outer belt at uniformly lower latitudes than the ∅4 orbit. While the dose encountered in one orbit of the ∅2 case is not excessive, it is high enough to preclude a multiple orbit mission.

Thc five orbits with the same insertion point and varying apogee altitudes reveal no correlation between dose and apogee altitude. In all five cases the dose per orbit was not excessive. In particular, only in cases 1 and 3 was the dose high enough to preclude missions involving a number of orbits.

7.2.5 Conclusions

The results obtained indicate that for the three classes of trajectories studied, the integrated dose absorbed in passing through the Van Allen belts will not constitute a hazard. In some instances a small variation in the controlled variables of a flight will be necessitated to reduce the dose to a very low value as in the emergency return, IIA and in the orbiting case, ∅2. The analyses, subject to the assumptions stated above, give a conservatively high estimate of the integrated dose. This serves to increase the confidence in the assertion that the safety of the crew would not be endangered by traversal of the trapped radiation belts in a normal lunar or orbital mission or in a prematurely terminated mission of either type.

7.3 SOLAR FLARE RADIATION

7.3.1 Introduction

The emissions of high intensity streams of energetic protons from the surface of the sun, termed solar flares, represent a potentially severe radiation hazard. At the present time limited data are available regarding the energy spectra and frequency of occurrence of these events. In the period from 1940 to the present direct and indirect measurements have been made of solar flares and associated phenomena. The data have provided the basis for relatively few concrete conclusions regarding the frequency of occurrence and possible correlations with other phenomena. The number of flares which have been measured directly is quite small and as a result the energy spectrum is not known for the vast majority of recorded events. Studies presently being conducted are aimed at establishing the existence of correlated variations in solar activity and the possibility of predicting major events. Present knowledge permits little more than the assertion that there is an eleven year periodicity to the phenomenon, and an apparent peak frequency of occurrence of approximately twelve high intensity events per year. The study of solar flare radiation treats the subjects of radiation shielding and encounter probability, with analyses based on conservative estimates of flare characteristics and behavior.

7.3.2 Radiation Shielding

7.3.2.1 STRUCTURAL ANALYSIS

In order to establish the shielding requirement for the vehicles being studied, an analysis was undertaken to determine the effective shielding afforded by the vehicle structure and equipment. The vehicle structure was evaluated by considering the actual mass per unit area (gm/cm^2) of those portions of the structure which enclosed the entire crew compartment. These included ablative coatings of the re-entry vehicle and walls of the pressure vessel. For purposes of this analysis, no distinction was made between different materials. This is equivalent to assuming that the stopping power of a material is independent of atomic number. The error introduced by this simplifying assumption is small when considering protons whose energies are in the region of interest (Sternheimer). Figure I-7-14 illustrates a structural detail of the D-2 re-entry vehicle and the values of shielding obtained in two areas of the structure. Applying a similar analysis to the total vehicle configuration yields a vehicle mass distribution plot as shown in Figure I-7-15 for the D-2 and in Figure I-7-16 for the R-3 vehicle. In the case of the D-2 axial symmetry obtains and the three-diminsional plot may be considered to be a revolution of the one shown in Figure I-7-15. For the R-3 the structure was divided into an upper and lower region with different construction. For this case, the three-dimensional equivalent would be obtained by rotating each half of the mass distribution through 180 degrees. In the mass distribution plot, the radius is equal to the shielding thickness in gm/cm^2, and consequently circles are loci of constant shielding. Using these plots, it is possible to determine for a given value

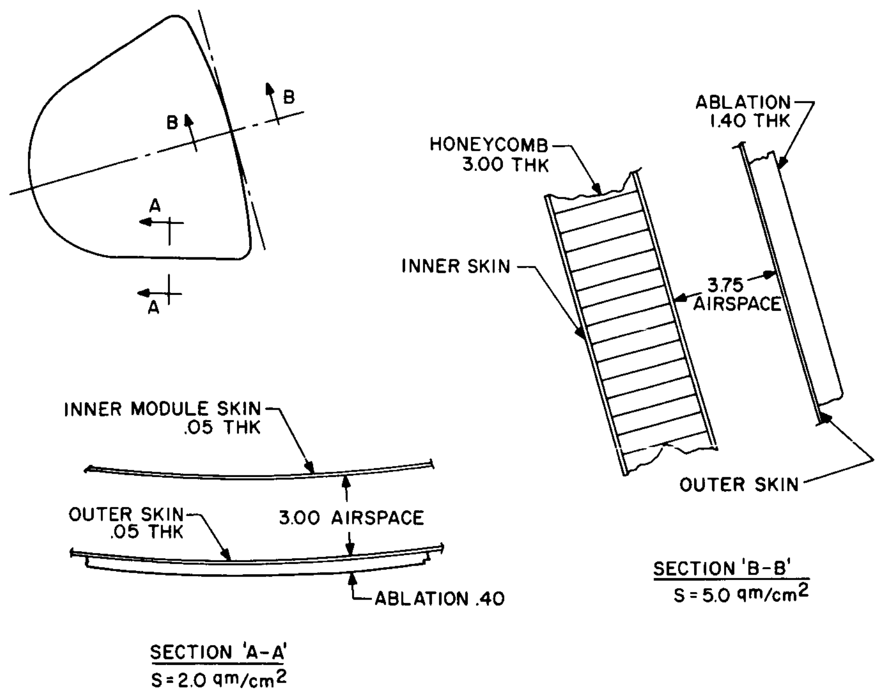


Figure I-7-14. Shielding analysis structural detail D-2 R/V

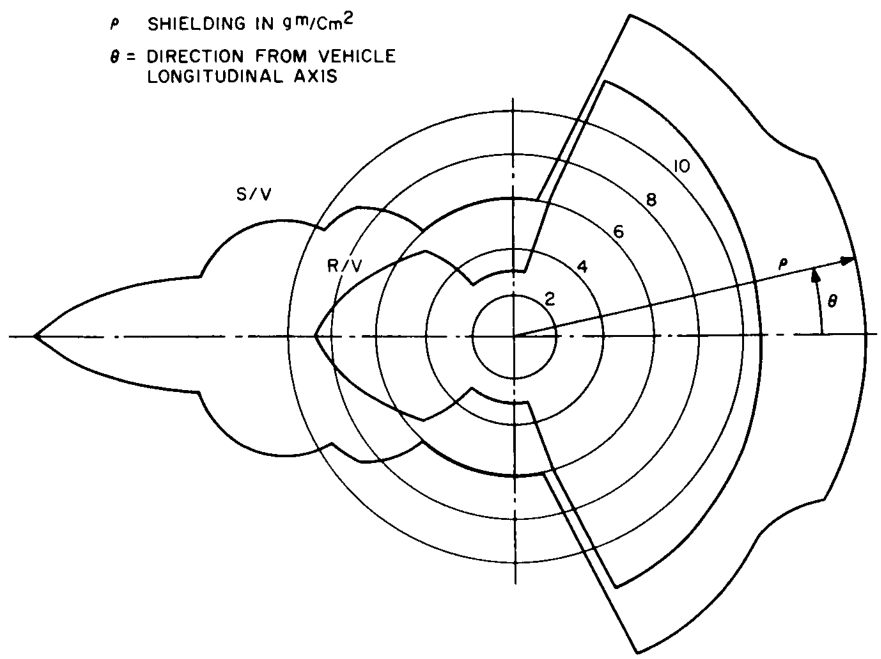


Figure I-7-15. Vehicle mass distribution D-2 configuration

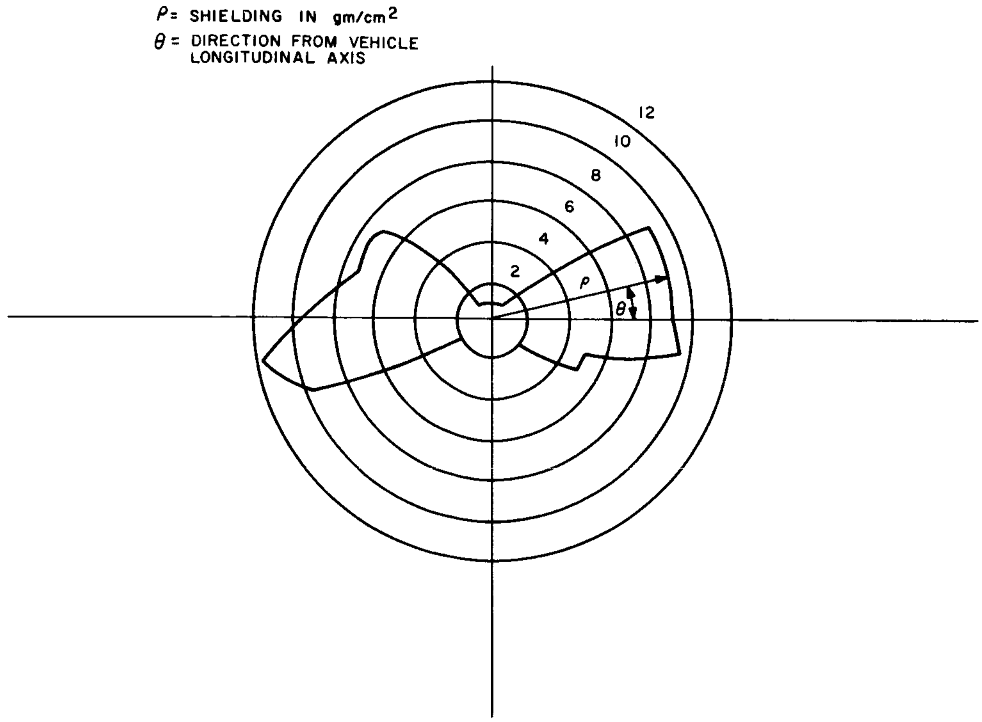


Figure I-7-16. Vehicle mass distribution R-3 configuration

of shielding, S, the solid angle subtended by the portion of the vehicle having less than the given value of shielding. This is termed the vulnerable solid angle, Θ_V. For values of S for which the circle with radius S lies entirely within the vehicle contour, the vulnerable solid angle Θ_V is equal to zero. For values of S for which the vehicle contour lies entirely within the circle of radius S, Θ_V is equal to 4π. If the locus of constant shielding thickness intersects the vehicle contour at angle Θ, and an arc of the circle equal to ψ degrees lies outside the contour, then the vulnerable angle is given by:

$$\Theta_V = 2\pi \left[\cos\Theta - \cos(\Theta + \psi)\right] \tag{1}$$

The relationships Θ_V as a function of S for the D-2 and R-3 vehicles are shown in Figures I-7-17 and I-7-18. This relationship characterizes completely the shielding properties of a given vehicle, and provides the basis for an extensive analysis of the shielding problem.

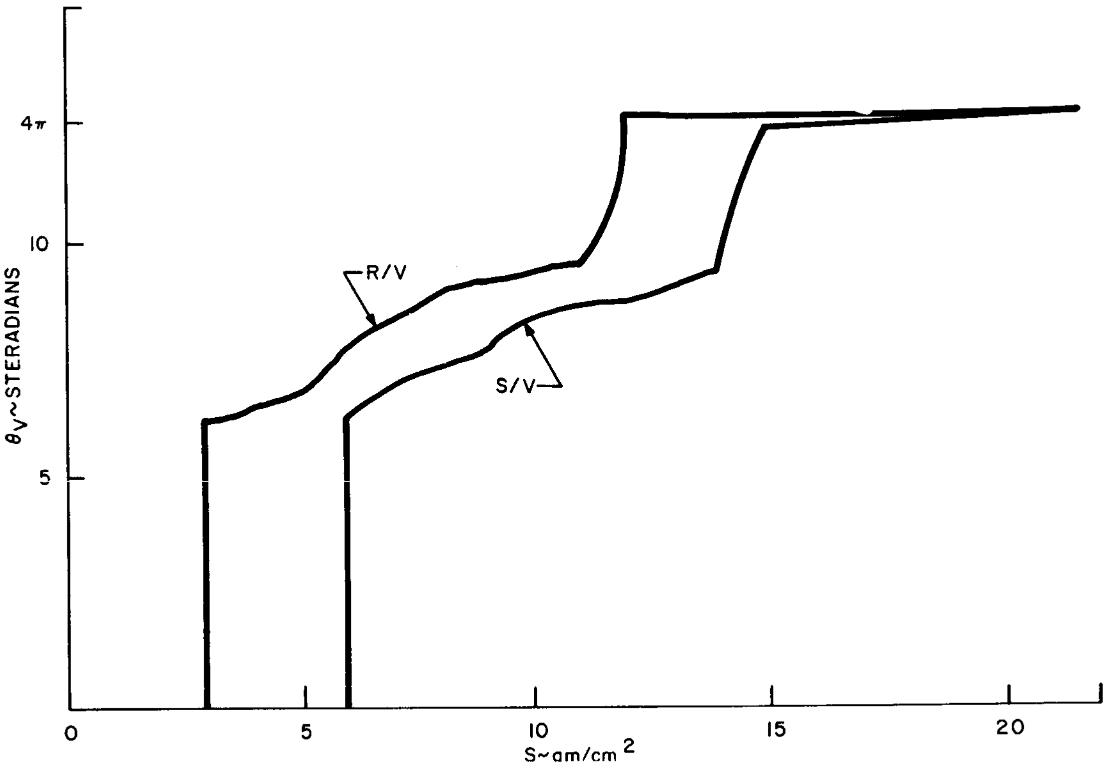


Figure I-7-17. Vulnerable solid angle vs. shielding thickness D-2

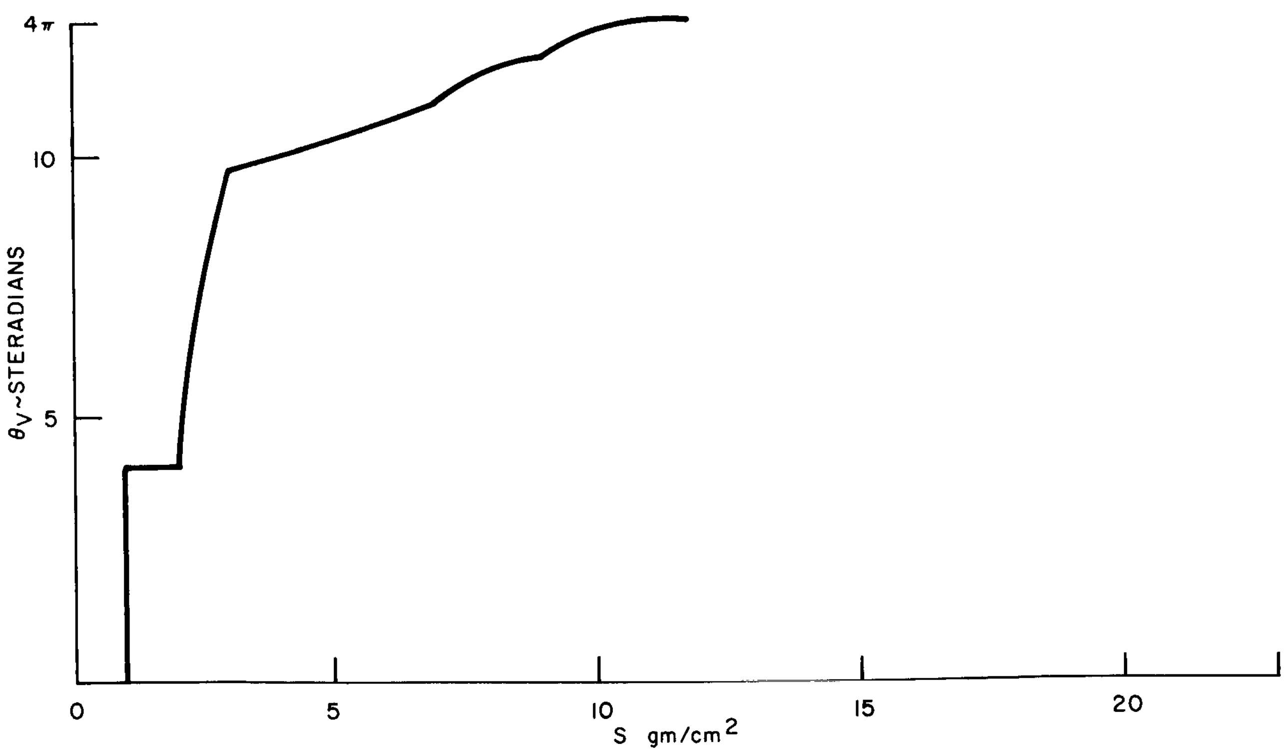


Figure I-7-18. Vulnerable solid angle vs. shielding thickness R-3

7.3.2.2 SUPPLEMENTAL SHIELDING WEIGHT

In order to evaluate the weight penalty associated with a given value of total vehicle shielding, it is necessary to determine the supplemental weight required to increase the vehicle shielding in vulnerable areas. To facilitate this computation, it is assumed that the additional mass added to the structure will be in the form of segments of a sphere of seven foot diameter. To achieve a given value of shielding, K gm/cm^2, each increment of area possessing mass, S, must be supplemented by (K-S) gm/cm^2. Expressed in equation form this yields:

$$\Delta W = r^2 (K-S) \Delta\Theta \tag{2}$$

Summing the above for all values of S less than K and passing to the limit yields:

$$W_{supp.} = r^2 \int_{\Theta(S_{min})}^{\Theta(K)} (K-S)\, d\Theta \tag{3}$$

In this expression, S is implicitly taken as a function of Θ, the inverse of the relationship plotted in Figures I-7-17 and I-7-18. The expression for W_{supp} was evaluated as a function of K, minimum shielding thickness, and the results are shown in Figure I-7-19. It may be seen that the weight penalty incurred in supplemental shielding is uniformly lower for the D-2 re-entry vehicle than for the R-3. When considering the D-2 in the space vehicle configuration, further decrease in weight penalty is achieved. However, even for this configuration, the supplemental shielding weights rapidly become impractical for values of K in excess of 10 gms/cm^2. The consideration of solar flare dose as a function of available shielding will resolve the requirement for supplemental shielding.

7.3.2.3 SOLAR FLARE DOSE CHARACTERISTICS

At this time the energy spectrum and intensities of only a relatively few class 3+ solar flares are known well enough to permit calculation of the dose vs. shielding characteristics. Foelsche and others have determined estimates for five flares.

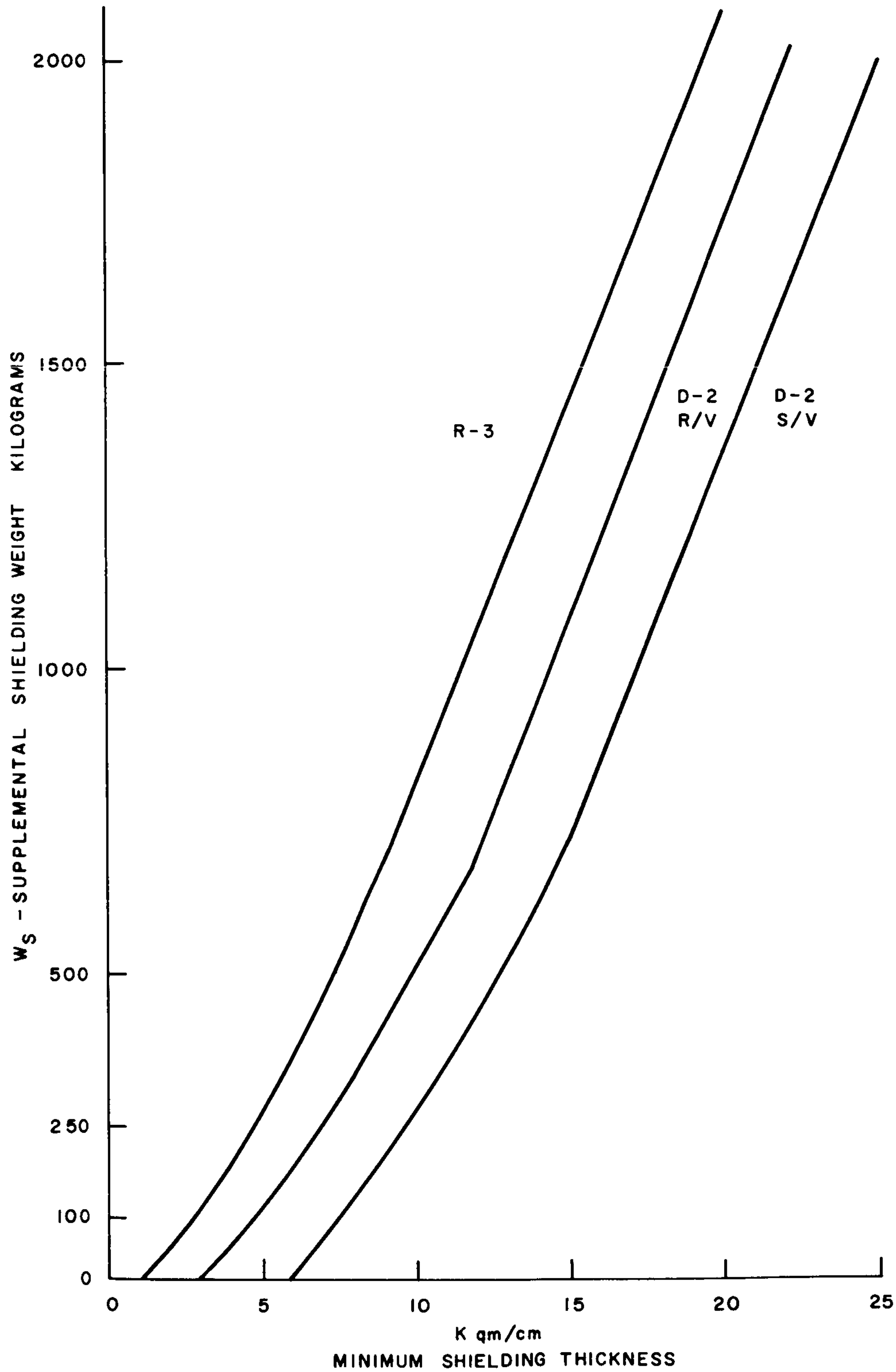


Figure I-7-19. Supplemental shielding weight vs. minimum shielding thickness

The characteristics may be classified in three categories according to the energy spectrum of the flare:

1. high intensity - low energy
2. extreme intensity - low energy
3. high intensity - high energy

The events of March 23, 1958 and August 22, 1958 are classed in category one. The events of May 10, 1959 and July 14, 1959 are in category two, and category three applies to events which produce changes in cosmic ray flux at sea level, represented by the flare of February 23, 1956. The latter type has been observed five times since 1942 and due to its infrequent occurrence is not considered from the point of view of dose characteristic. Figure I-7-20 shows the dose vs shielding characteristics for flares in the three categories. Approximate equations for the curves of August 58 and May 59 are given below. Shielding, S, is in gm/cm^2 of H_2O.

August 58 $$D = \frac{515}{S^{3.1}} \text{ Rem} \tag{4}$$

May 59 $$D = \frac{16.7 \times 10^4}{S^{2.74}} \text{ Rem} \tag{5}$$

7.3.2.4 WEIGHT PENALTY VS. ALLOWABLE DOSE

The flare dose characteristics (Figure I-7-20) and vehicle supplemental weight characteristic (Figure I-7-19) provide the necessary information for establishing the weight penalty required to limit the dose from a given flare to a predetermined allowable dose. Table I-7-V summarizes the shielding required to limit dose for the three flare types. Table I-7-VI gives the supplemental shielding weight for the R-3 and D-2 vehicles resulting from the shielding requirements of Table I-7-V. Examining the results for the August 58 type flare it is seen that no supplemental shielding is required for the D-2 configuration to limit the dose to 25 rem; for the R-3, 100 kg of supplemental weight is required. Considering the flares of the second and third category, it becomes evident that the shielding weights are impractically high even

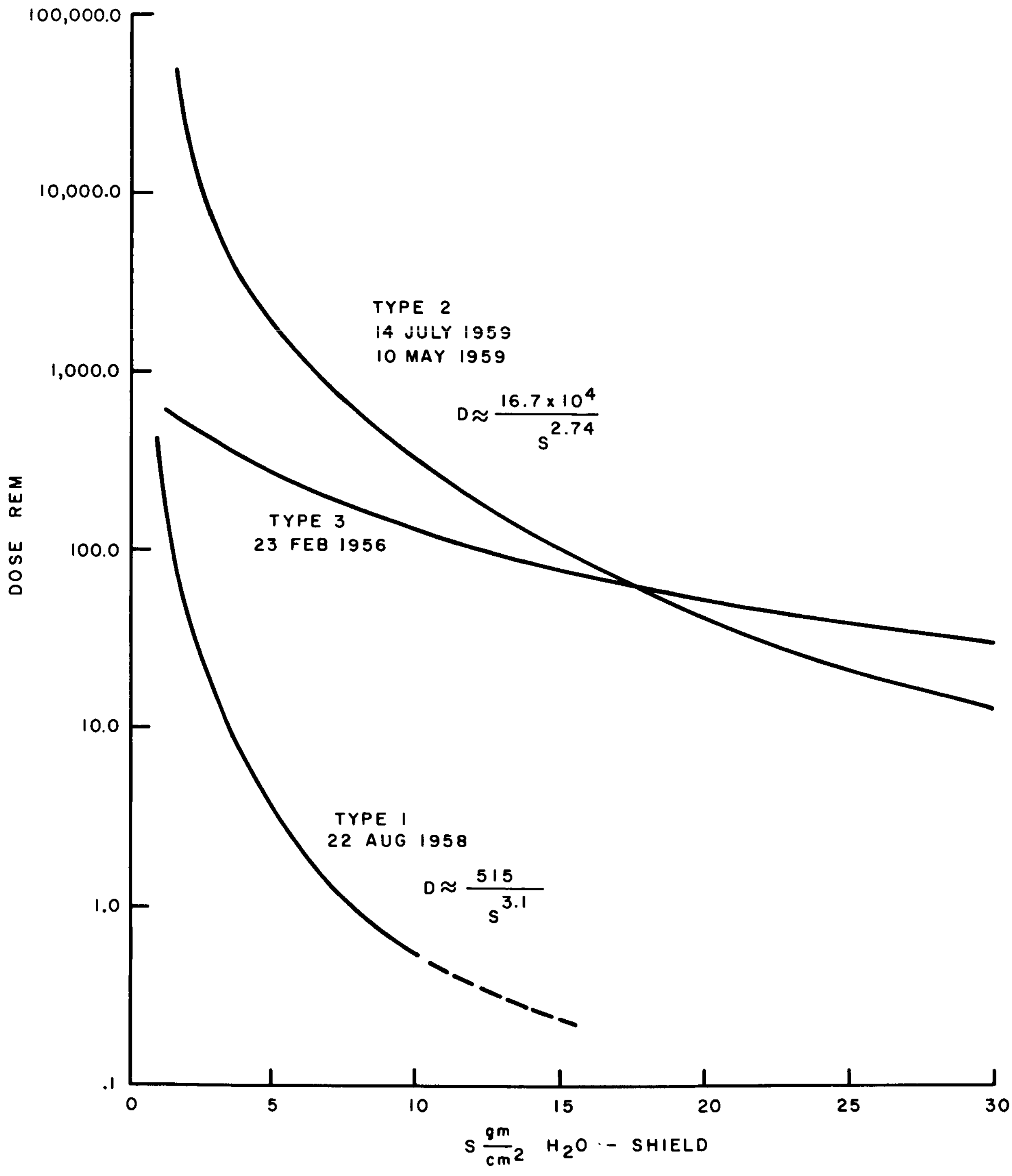


Figure I-7-20. Dose vs. shielding for two flares (Foelsche)

TABLE I-7-V. SHIELDING REQUIRED TO LIMIT DOSE
$S \sim gm/cm^2\ H_2O$

Dose Rem	Flare		
	Aug. '58	May '59	Feb. '56
1	8 gm/cm^2	—	—
5	4.2	—	—
10	3.5	34	—
15	3.3	28	—
20	3.0	27	36
25	2.8	25	32
50	2.5	19	21
100	2.0	15	13

TABLE I-7-VI. SUPPLEMENTAL SHIELD WEIGHT TO LIMIT DOSE
$W \sim$ KILOGRAMS

Dose Rem	Aug. '58			May '59			Feb. '56		
	R-3	D-2 R/V	D-2 S/V	R-3	D-2 R/V	D-2 S/V	R-3	D-2 R/V	D-2 S/V
1	675	335	100	W > 2,000 Kg			W > 2,000 Kg		
5	180	50	0	W > 2,000 Kg			W > 2,000 Kg		
10	150	30	0	W > 2,000 Kg			W > 2,000 Kg		
15	125	15	0	W > 2,000 Kg			W > 2,000 Kg		
20	115	0	0	W > 2,000 Kg			W > 2,000 Kg		
25	100	0	0	W > 2,000 Kg			W > 2,000 Kg		
50	75	0	0	1950	1580	1210	2200	1850	1450
100	50	0	0	1440	1080	730	1125	750	475

when the allowable dose is increased to 100 rem. This result obtains for all vehicles considered and indicates that a low probability of encountering such a flare is the only available means of mitigating the associated hazard.

7.3.2.5 FLARE EQUIVALENT SHIELDING

An analysis was made to develop a method of characterizing a given vehicle by an effective value of shielding. A strict physical evaluation of the average thickness would not be of significance due to the non-linear relationship between dose and shielding. A valid estimate of the equivalent shielding may be derived by determining the actual dose produced in the vehicle by a given flare. For this purpose, the flare dose characteristic and vehicle shielding characteristic are required. The total dose absorbed in the vehicle is given by:

$$D_T = \frac{1}{4\pi} \int_0^{4\pi} D(S)\, d\Theta_v = \frac{1}{4\pi} \int_{S_{min}}^{S_{max}} D(S) \left(\frac{d\Theta_v}{ds}\right) (ds) \tag{6}$$

This expression was evaluated for the R-3 and D-2 vehicles using the vulnerability curves of Figure I-7-17 and I-7-18, and the two dose characteristics of Figure I-7-20. For computational purposes the vehicle shielding characteristics were represented by a function composed of a step rise at the minimum value of shielding, and two straight line segments.

$$\Theta_v = 0 \quad \text{where } 0 \leq S \leq S_{min} \tag{7a}$$

$$\Theta_v = \Theta_1 + M_1 (S - S_{min}) \quad \text{where } S_{min} \leq S \leq S_2 \tag{7b}$$

$$\Theta_v = \Theta_2 + M_2 (S - S_2) \quad \text{where } S_2 \leq S \leq S_{max} \tag{7c}$$

Incorporating the approximate equations for the flare dose characteristics:

$$D = \frac{B}{S^K}$$ yields the following:

$$D_T = \frac{B}{4\pi}\left\{\frac{\Theta_1}{(S_{min})^k} + \frac{M_1}{K-1}\left(\frac{1}{S_{min}^{\;K-1}} - \frac{1}{S_2^{\;K-1}}\right) + \frac{M_2}{K-1}\left(\frac{1}{S_2^{\;K-1}} - \frac{1}{S_{max}^{\;K-1}}\right)\right\} \tag{8}$$

The values of dose and effective values of shielding resulting therefrom appear in Table I-7-VII.

TABLE I-7-VII. FLARE EQUIVALENT SHIELDING

	Dose - Aug. 58	Dose - May 59	Effective Shielding (gm/cm^2)
R-3	21.4 REM	7.4×10^4 REM	2.7 - 2.8
D-2 R/V	16.3	5.3×10^3	3.0 - 3.3
D-2 S/V	1.2	7.4×10^2	7.0 - 7.3

The data in Table I-7-VII reinforce two earlier conclusions. The first is the evident superiority of the shielding afforded by the D-2 in the space vehicle configuration. The second is the assertion that only flares of the type of May 59 will constitute a clear radiation hazard.

7.3.3 Encounter Probability

7.3.3.1 ASSUMPTIONS AND METHOD

The probability of encountering one or more solar flares on a given mission is naturally dependent on the duration of the mission and the frequency of occurrence of the solar events. Secondarily, the duration of each event, and any temporal variation or correlation of their occurrence will influence the probability of encounter. Prediction of the occurrence offers a means of reducing the encounter probability, when that information is used as a basis for determining a permissible launch date or conditions necessitating emergency termination of the mission.

While the existence of some correlation in the solar activity would by aiding prediction decrease the probability of encounter, none has been satisfactorily established to date.

Therefore, the analysis which follows will treat the events as being random and uncorrelated. The duration of a solar flare may be as great as 200 hours. However, due to the rapid decrease of intensity with time the effective duration is generally under 20 hours. The events may then be treated as being effectively of one day's duration.

In the discussion which follows the following notation is employed:

EP = Encounter Probability

f = Flare Frequency/year

t = Mission Duration - Days

p = Prediction Period - Days

q = f/365, one day Encounter Probability

$$\binom{n}{r} = \frac{n!}{(n-r)!\ r!} = \text{Binominal coefficient}$$

The probability of encountering one or more flares occurring with frequency f, on a mission of duration t is given by:

$$EP = 1 - (1 - q)^t \tag{9}$$

Graphs of this function appear in Figure I-7-21 for f = 4, 6, and 12. Considering a year of peak activity, f = 12, the probability of encounter is equal to 37.4 percent. This represents the sum of the probabilities of encountering 1, 2.......... up to 12 flares. The probability of encountering precisely one event is given by:

$$EP_1 = tq\,(1-q)^{t-1} \tag{10}$$

This relationship is shown in Figure I-7-22 for f = 12. The value for a 14 day mission is 28.8 percent. Thus, it is seen that the cases in which multiple encounters take place occur in 8.6 percent of two week missions. The significance of multiple encounters depends upon the types of flares which occur and their relative frequency of occurrence and are discussed below.

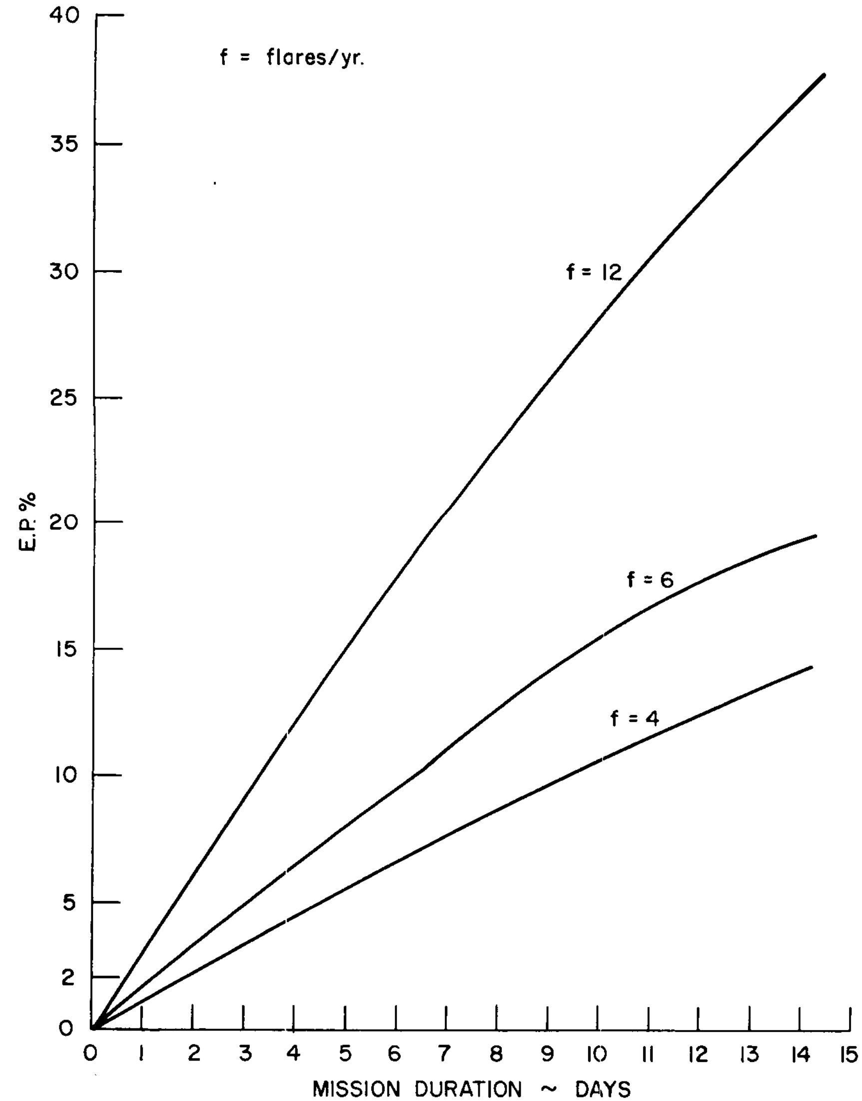


Figure I-7-21. Encounter Probability vs mission duration

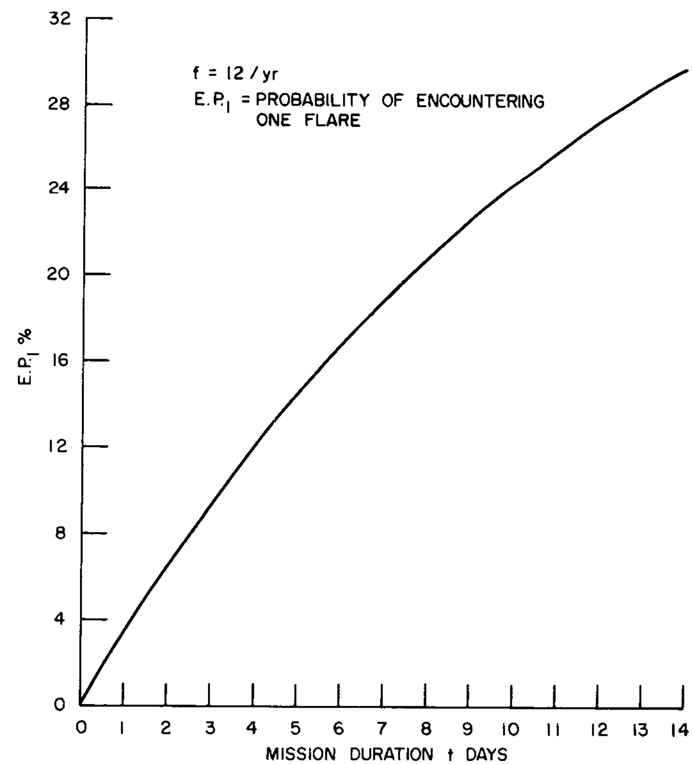


Figure I-7-22. Encounter probability vs. mission duration

7.3.3.2 FREQUENCY - INTENSITY MODELS

The frequency of high intensity solar flares determines the various probabilities which are of interest. However, the relative occurrence of flares of various types determines the actual hazard represented by the possible modes of encounter. For this purpose a number of models were selected as being possible representations of the relationship between frequency of occurrence and intensity or flare type. The models employ only the type 1 and type 2 flares described in paragraph 2, above. In each model a fraction of the total flares occurring is assigned to each type. Cases in which the total number of events are twelve and six are discussed, representing a year of peak activity and one of moderate activity respectively. The probabilities of encountering one flare of either type and the probabilities of encountering two flares of one or both types is tabulated in Table I-7-VIII. The pair encounter probability is given by:

$$EP_2 = \binom{t}{2} q^2 (1 - q)^{t-2} \tag{11}$$

TABLE I-7-VIII. FREQUENCY - INTENSITY MODELS

Model	f_a	f_m	EP_1	EP_A	EP_M	EP_2	EP_{AA}	EP_{AM}	EP_{MM}
I	10	2	29.8	24.8	5.0	6.6	4.5	2.0	0.1
II	9	3	29.8	22.4	7.4	6.6	3.6	2.7	0.3
III	8	4	29.8	19.9	9.9	6.6	2.8	3.2	0.6
IV	3	3	18.6	9.3	9.3	2.0	0.4	1.2	0.4
V	4	2	18.6	12.4	6.2	2.0	0.8	1.1	0.1
VI	5	1	18.6	15.5	3.1	2.0	1.3	0.7	0.0

For a particular intensity-frequency model the number of type 1 flares is denoted f_A (August 58) and the type 2 by f_M (May 59). Three modes of pair encounter are possible:

EP_{AA} = encounter of two of the August 58 type

EP_{MM} = encounter of two of the May 59 type

EP_{AM} = encounter of one of each type

[redacted]

The following equations obtain for the three cases:

$$EP_{AA} = \frac{\binom{fa}{2}}{\binom{f}{2}} EP_2 \tag{12a}$$

$$EP_{MM} = \frac{\binom{fm}{2}}{\binom{f}{2}} EP_2 \tag{12b}$$

$$EP_{AM} = \frac{fmfa}{\binom{f}{2}} EP_2 \tag{12c}$$

Graphs of equations 12a-c appear in Figure I-7-23 for one intensity-frequency model.

7.3.3.3 EFFECT OF PREDICTION

The decrease in risk afforded by the ability to predict may be represented by the ratio of encounter probability with prediction to encounter probability with no prediction. It is easily seen that this quantity will vary between one and zero for prediction periods ranging between zero and the total mission duration.

This ratio is given by:

$$\frac{EP_k(P)}{EP_k(O)} = \frac{\binom{t-p}{k} q^k \ (1-q)^{t-k-p}}{\binom{t}{k} q^k \ (1-q)^{t-k}} \tag{13}$$

where K is equal to the number of encounters.

Evaluating this expression for a 14 day mission and probabilities of one encounter (K=1) yields:

$$\frac{EP_1 \ (P)}{EP_1 \ (O)} = \left(1 - \frac{P}{14}\right)\left(1 - q\right)^{-P} \tag{14}$$

[redacted]

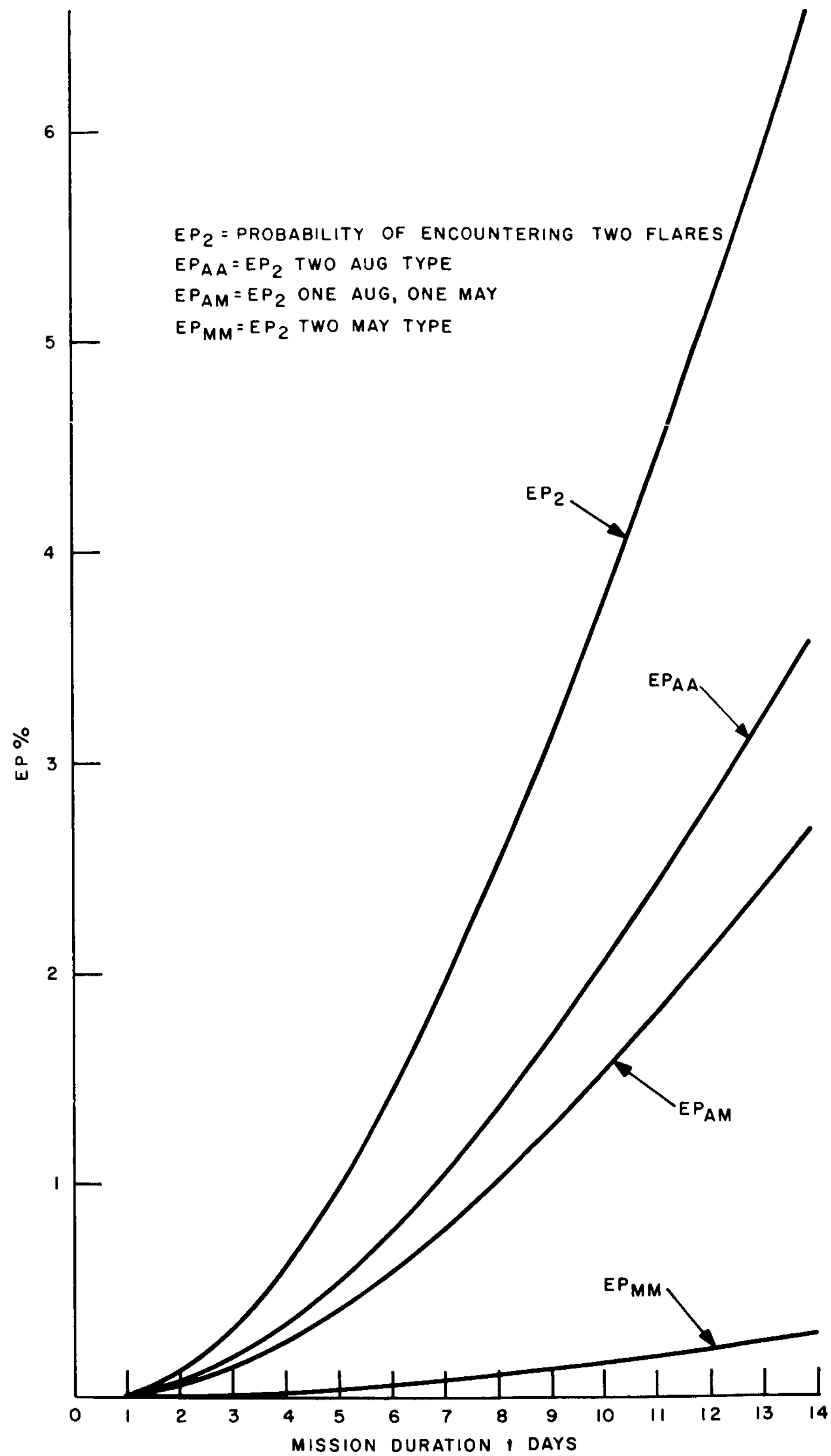


Figure I-7-23. Pair encounter probability vs. mission duration

[redacted]

This expression is plotted in Figure I-7-24 for three values of q. Risk reduction of only 1/10 can be achieved with a prediction period of 2 days. In order to reduce the risk to one half the value without prediction, more than a seven day prediction period is required.

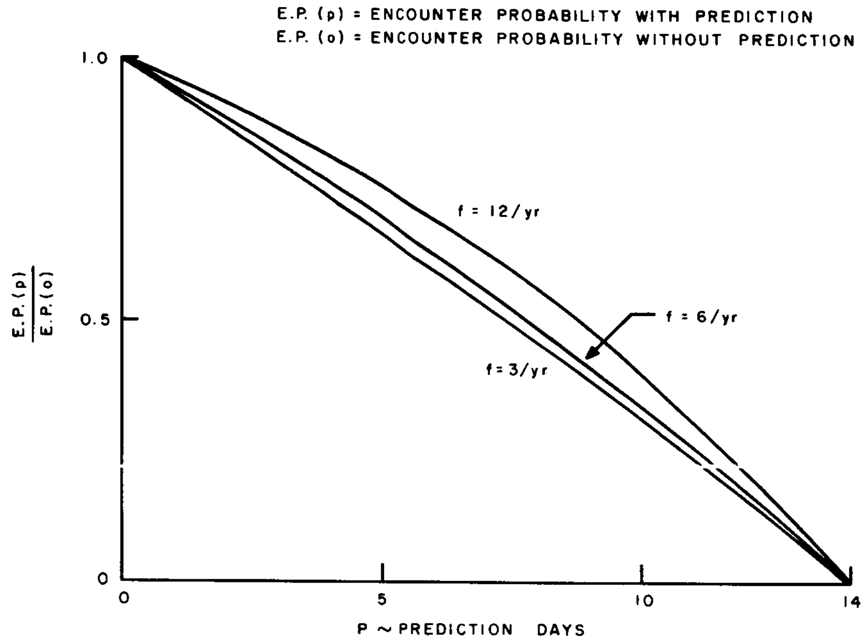


Figure I-7-24. Effect of prediction $\frac{E.P_{(p)}}{E.P_{(o)}}$ vs. P

7.4 SECONDARY RADIATION

7.4.1 Introduction

The interaction of high-energy protons with material can produce a number of different types of secondary products. The types with which we are principally concerned are neutrons and gamma rays, because of their ability to penetrate the wall material between the inside surface and the point of their birth.

The dose rates are calculated at the center of a spherical shell and with the assumptions that the secondary products are emitted isotropically and are not absorbed by the wall.

Under these special conditions, the flux density of neutrons or gammas at the center of the sphere is equal to their birth rate per unit area of the surface of the sphere.

7.4.2 Neutrons

A derivation and statement of the equations for calculating the dose rate due to neutrons produced by proton bombardment follows.

If σ is the cross-section for absorption of a proton of a given energy by a certain material, and n is the average number of neutrons produced by this absorption, the product σn is the neutron production cross-section. Measured values of σn are given by Crandall & Millburn for 300-Mev protons and by Tai, Millburn, Kaplan and Moyer for protons of 0 to 18 Mev and of 0 to 32 Mev. Thses values are plotted in Figure I-7-25.

The data of Millburn et al are given in terms of neutron yields, and they are changed to neutron production cross section by means of the relation:

$$Y = \overline{\sigma n} \frac{N_o P_n}{A} \rho x \qquad (15)$$

where Y = yield of neutrons (thick target),

$\overline{\sigma n}$ = average value of σn over the energy interval zero to the initial proton energy, in cm^2,

N_o = Avagodro's number,

P_n = number of protons hitting target,

A = atomic weight of target material

ρ = density of target material, in g/cm^3, and

x = range of proton, in cm.

In order to obtain an expression for σn as a function of proton energy, a plot of these two variables is made and a power law curve fitted to the points, as shown in Figure I-7-26 for aluminum and Figure I-7-27 for cadmium. The line from zero to 18 Mev

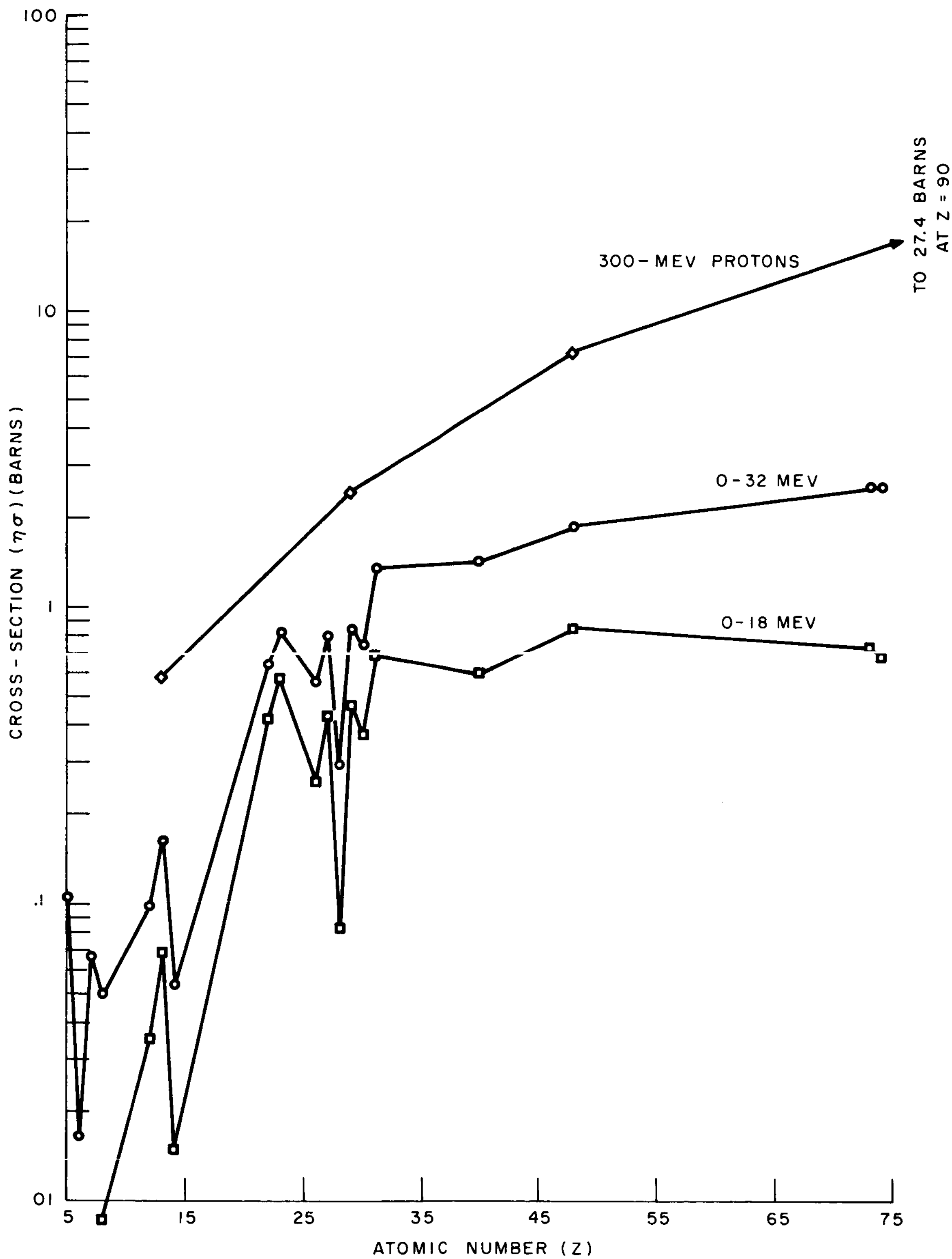


Figure I-7-25. Neutron-production cross-section vs. atomic number of target for protons of various energies

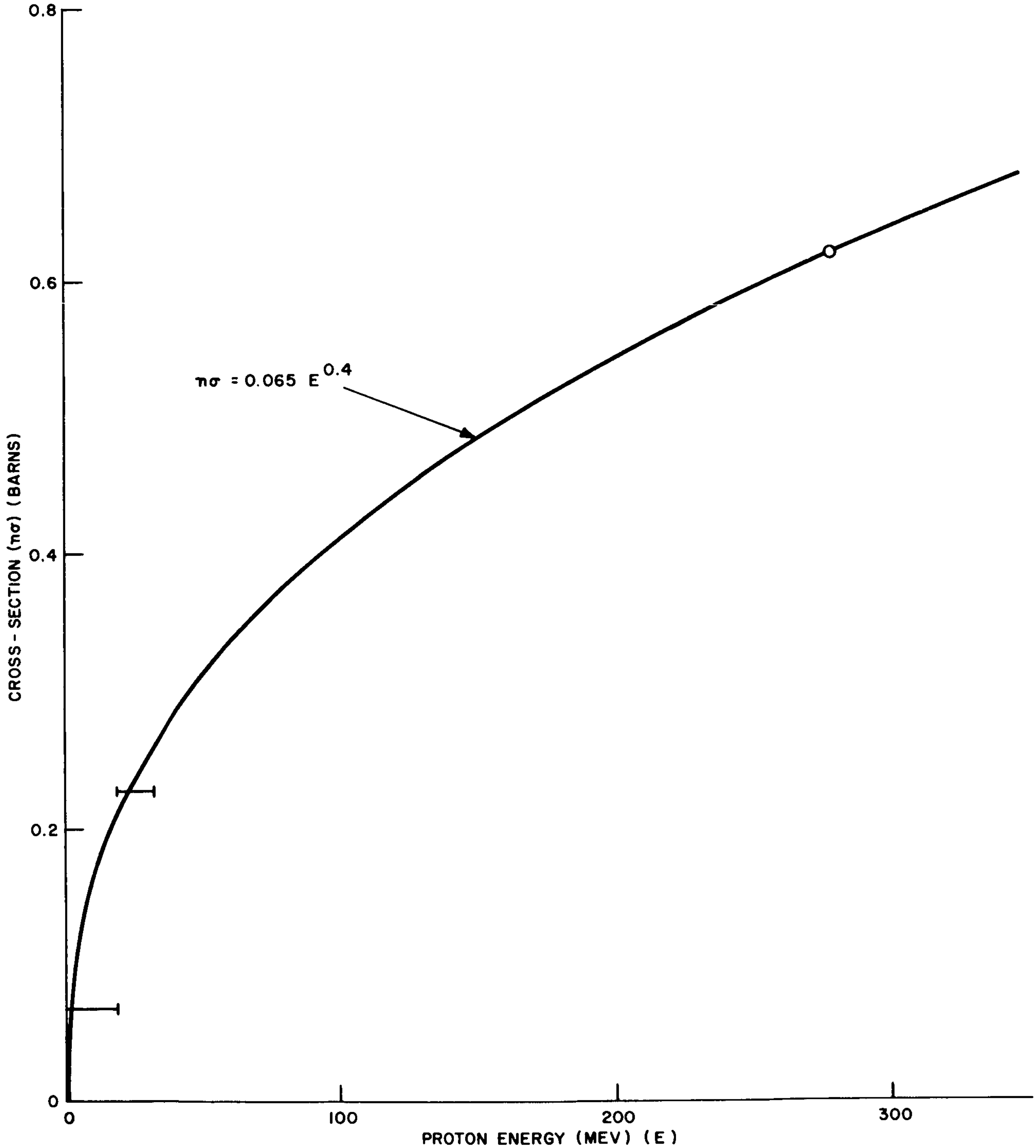


Figure I-7-26. Neutron-production cross-section vs. proton energy for aluminum target

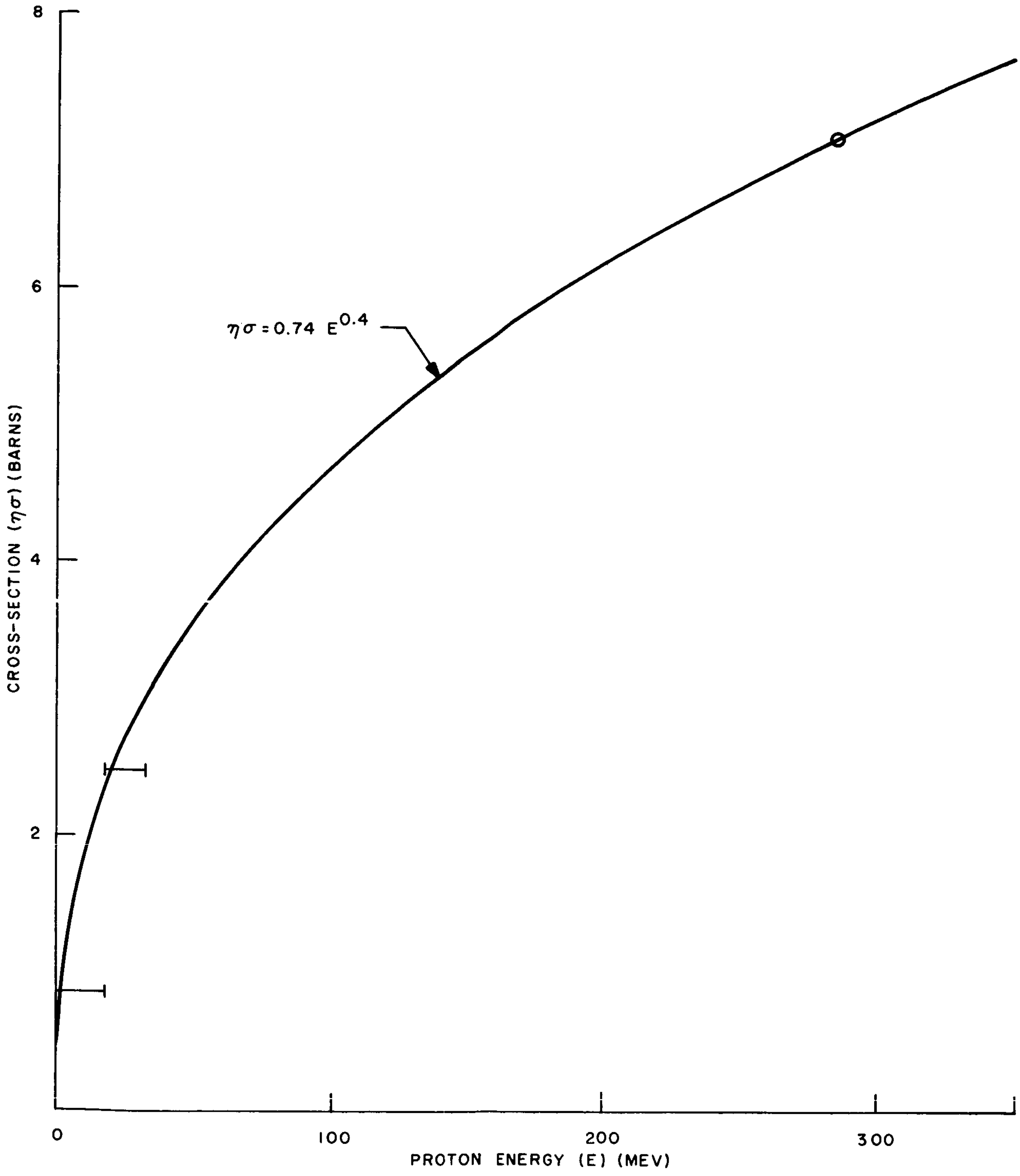


Figure I-7-27. Neutron-production cross-section vs. proton energy for cadmium target is the average cross-section over this range, and the line from 18 to 32 Mev is the average cross-section over this range. This latter value was obtained from the 0 to 18-Mev and 0 to 32-Mev values by means of the following expression:

$$Y_{32} = Y_{18} + \overline{\sigma n}_{18-32} \frac{\text{No Pn}}{A} \left(X_{32} - X_{18} \right) \tag{16}$$

where Y_{32} = thick target yield of neutrons from 32-Mev protons,

Y_{18} = thick-target yield of neutrons from 18-Mev protons,

$\overline{\sigma n}_{18-32}$ = average value of σn over the energy range 18 to 32 Mev,

x_{32} = range of 32-Mev protons,

x_{18} = range of 18-Mev protons.

The expressions so obtained are

$$\sigma n = 0.065 E^{0.4} \tag{17}$$

for aluminum, and

$$\sigma n = 0.74 E^{0.4} \tag{18}$$

for cadmium, where

σn = neutron-production cross-section, in barns (1 barn = $10^{-24} cm^2$), and

E = proton energy, in Mev.

For aluminum, the number of neutrons of all energies generated per unit path length by a proton of energy E is given by:

$$dn = \frac{No}{A} (0.065 \times 10^{-24} E^{0.4}) \rho \, dx \tag{19}$$

and this can be written as:

$$dn = \frac{No}{A} \left(0.065 \times 10^{-24} E^{0.4} \right) \rho \frac{dx}{de} \, dE \tag{20}$$

To obtain an expression for $\rho \frac{dx}{de}$, we write the range of protons in aluminum as:

$$E_o = 27 (\rho R)^{0.565} \tag{21}$$

where E_o = initial energy of proton, in Mev,

ρ = density of aluminum, in g/cm^3,

R = range of proton, in cm.

When this proton has penetrated a distance x, its residual range is R-x, and its remaining energy is, from (21),

$$E = 27 (\rho R - \rho x)^{0.565} \tag{22}$$

Rearrangement and differentiation of (22) yields:

$$\rho \frac{dx}{de} = - \frac{1}{.565 \times 27} \left(\frac{E}{27}\right)^{\frac{.435}{.565}} \tag{23}$$

Substituting (23) into (20) and integrating between the initial energy of the proton, E_o, and its final energy, zero, we obtain the yield of neutrons in aluminum by a proton of energy E_o, when this proton is completely stopped in the wall.

$$n = 3.48 \times 10^{-6} (E_o)^{2.17} \tag{24}$$

We must also consider the production of neutrons by protons that do not stop in the wall. This is done by integrating (19) from zero to T, the wall thickness, using a constant value for the cross-section namely its maximum value at the initial proton energy, E_o. Doing this, we obtain for the yield of neutrons in aluminum by a proton of energy E_o, when this proton is not stopped in the wall,

$$n = 1.45 \times 10^{-3} E^{0.4} \rho T \tag{25}$$

If the differential flux density of protons in the environment is f (E_o) protons cm^{-2} sec^{-1}, and if they are isotropic, the number of protons of energy E_o to $E_o + dE_o$ striking unit area of the vehicle wall is

$$dN_P = \frac{1}{4} f(E_o) \; dE_o \tag{26}$$

The neutron flux density at the center of the sphere, due to these protons, is:

$$d\phi = \frac{1}{4} \times 3.48 \times 10^{-6} (E_o)^{2.17} f(E_o) \, d\, E_o \tag{27}$$

for protons that stop in the wall, or

$$d\phi = \frac{1}{4} \times 1.45 \times 10^{-3} \rho T \, (E_o)^{0.4} f(E_o) \, d\, E_o \tag{28}$$

for protons that penetrate the wall.

The total neutron flux density is obtained by integrating (27) from the lowest value of proton energy to E_R, the energy of a proton that can just penetrate the wall, and integrating (28) from E_R to the highest value of proton energy, and adding the results. This yields

$$\phi = \frac{1}{4} \times 3.48 \times 10^{-6} \int_{E_1}^{E_R} (E_o)^{2.17} f(E_o) \, d\, E_o$$

$$+ \frac{1}{4} \times 1.45 \times 10^{-3} \rho T \int_{E_R}^{E_2} (E_o)^{0.4} f(E_o) \, d\, E_o \tag{29}$$

Now for neutrons between 1 and 10 Mev, a dose rate of 0.1 rem per 40 hours is produced by a flux density of 17 to 20 neutrons $\mathrm{cm}^{-2} \, \mathrm{sec}^{-1}$.

This is equivalent to saying that one neutron $\mathrm{cm}^{-2}\mathrm{sec}^{-1}$ produces a dose rate of $4\mathrm{x}10^{-8}$ rem sec^{-1}. Multiplying equation (29) by this value gives the expression for dose rate at the center of the spherical vehicle.

$$d = 3.48 \times 10^{-14} \int^{E_R} (E_o)^{2.17} f(E_o) \, d\, E_o +$$

$$1.45 \times 10^{-11} \rho T \int_{E_R}^{E_2} (E_o)^{0.4} f(E_o) \, d\, E_o$$

where d = dose rate, in rems sec^{-1}

E_1 = energy, in Mev, of lowest energy proton,

E_2 = energy, in Mev, of highest energy proton,

E_R = energy, in Mev, of proton whose range equals the shell thickness,

$f(E_o)$ = differential omnidirectional flux density of protons, in cm^{-2} sec^{-1} Mev^{-1}, and

T = thickness of shell, in g cm^{-2}.

It remains to demonstrate that the majority of neutrons produced have energies between 1 and 10 Mev, for this is the assumption used above. We will use the information that the neutron energy spectrum has a (normalized) distribution given by:

$$N(E)\, dE = \frac{E}{K^2}\, e^{-E/K}\, dE \tag{31}$$

and that the mean energy of the neutrons is between 2 and 5 Mev when the proton energy is between 300 and 820 Mev.

The mean value of (31) is obtained by multiplying it by E and integrating from zero to infinity. This turns out to be

$$\overline{E} = 2K \text{ or } K = \frac{\overline{E}}{2} \tag{32}$$

The fraction of neutrons with energy greater than 10 Mev for $\overline{E}$ = 5 Mev is:

$$\int_{10}^{\infty} \frac{E}{(2.5)^2}\, e^{-E/2.5}\, dE = 0.09$$

and the fraction of neutrons with energy less than 1 Mev for $\overline{E}$ = 2 Mev is:

$$\int_{o}^{1} \frac{E}{(1)^2}\, e^{-E/1}\, dE = 0.26$$

Thus we see that the number of neutrons with energy greater than 10 Mev, for which the dose per neutron is greater than the value we used, is very small. Also the number of neutrons with energy less than 1 Mev is not negligible, however the dose per neutron for these lower energy neutrons is less than the value we used.

If the differential omnidirectional proton flux density of the solar flare of 23 Feb 56 at its start, as given by Foelsche, is written as:

$$f(E_o) = 1.18 \times 10^5 \, E_o^{-1.25}, \quad 1 < E_o < 90 \text{ Mev}$$

$$f(E_o) = 1.34 \times 10^7 \, E_o^{-2}, \quad 90 < E_o < 2140$$

$$f(E_o) = 3.82 \times 10^{24} \, E_o^{-7}, \quad 2140 < E_o < \infty$$

and this expression substituted into (30), the resulting dose rate for a 30 gm/cm^2 shield is:

$$d = 5 \times 10^{-4} \text{ rem/sec} \tag{33}$$

If the differential omnidirectional flux density of protons in the heart of the inner belt is given the shape shown by Freden and White and amplitude shown by Foelsche it can be written as:

$$f(E_o) = 3.9 \times 10^3 \, E_o^{-0.72}, \quad 10 < E_o < 100 \text{ Mev}$$

$$f(E_o) = 1.43 \times 10^6 \, E_o^{-2}, \quad 100 < E_o < \infty$$

If this expression is substituted into (30), the resulting dose rate for a 30 gm/cm^2 shield is:

$$d = 4.3 \times 10^{-5} \text{ rem/sec} \tag{34}$$

7.4.3 Gamma Rays

In this section we will compare the dose production rate due to gammas to the dose production rate due to neutrons.

D'Arcy states that the average cross-section for absorption of protons by air over a wide energy range is 0.2 barn, and that, on the average, 4 Mev of gamma radiation is emitted per interaction. If we use 0.4 barn for aluminum, the energy flux density of gammas at the center of our spherical shell for 1 proton/cm^2-sec incident is

$$E_\gamma = 4\,\sigma \frac{N_o}{A} \rho\, x \qquad (35)$$

The dose produced per Mev cm^{-2} of gamma radiation is, (from Cronkhite), 4.6 x 10^{-10}rem, so the dose rate at the center of our sphere due to gammas is:

$$d_\gamma = 4\,\sigma \frac{N_o}{A} \rho x \times 4.6 \times 10^{-10} \qquad (36)$$

For neutrons, on the other hand, the average value of $n\sigma$ (Figure I-7-26) is about 0.5 barns, and if we assume 0.4 for the value of σ, then the average value of n is 1.25. So the number flux density of neutrons at the center of our spherical shell for 1 proton cm^{-2}sec^{-1} incident is:

$$N_n = 1.25\,\sigma \frac{N_o}{A} \rho x \qquad (37)$$

The dose produced per neutron cm^{-2} is 4 x 10^{-8} rem, so the dose rate at the center of our sphere due to neutrons is:

$$d_n = 1.25\,\sigma \frac{N_o}{A} \rho x \cdot 4 \cdot 10^{-8} \qquad (38)$$

The ratio of these two dose rates is obtained by dividing (36) by (24), and this is

$$\frac{d_\gamma}{d_n} = 0.037 \qquad (39)$$

showing that the dose rate due to gammas is substantially less than that due to neutrons.

7.4.4 Conclusions

The results of the foregoing analysis indicate that for the two proton spectra which were studied the dose due to secondary neutrons will be small compared to that due to the primary protons. Similarly, the dose from gamma rays will be an even smaller fraction of the neutron dose, and therefore negligible. For the Van Allen belt, the neutron dose rate is less than 5 percent of the primary proton rate. For the flare of February 23, 1956 the neutron dose is 6 percent of the proton dose. The gamma ray dose is approximately 3.7 percent of the neutron dose in both cases, and when compared to the dose due to protons, represents a correction of less than one half of one percent. It must also be noted that the high value of shielding employed in the analyses will give an upper estimate to the rate of production of secondary particles. Therefore, for the vehicles under consideration it may be asserted that the analyses based on primary proton dose/rates are well applicable and that consideration of secondary radiation leads to no alteration of conclusions arrived at on that basis.

7.5 PROBABLE DOSE

The analyses of shielding, encounter probability and secondary radiation provide the basis for a determination of the probability distribution of the dose for a given vehicle. This may be determined by considering all modes of solar flare encounter, the associated probabilities, and the dose arising from each encounter. For this analysis two models of the frequency-intensity relationship are employed. Model II is chosen as a conservative estimate of the relationship in a year of peak solar activity, perhaps 1968. With a total of 12 high intensity flares occurring, three are of the type of May 1959 and the remainder one of the type of August 1958. Model VI is an estimate of the behavior during a year of moderate activity, perhaps 1971, in which six high intensity events occur, one of which is of the type of May 59. Values of encounter probability are taken from Table I-7-VIII. The probabilities are modified in accord with Equation 13 to include the effect of a two-day prediction capability.

The modes of encounter considered include single encounters, and all possible pair encounters. The probabilities of triple or higher order encounters are negligible. The doses arising in each case depend upon the shielding of the vehicle. The D-2 vehicle in the space vehicle configuration is characterized by flare-equivalent shielding of 7.0 gms/cm^2. This value is assumed in the probable dose analysis. Since the modes of encounter having significant probabilities are finite in number a continuous probability density distribution cannot be constructed. However, an integral distribution representing the probability of exceeding a given value of dose may be determined by summation. The probability of exceeding dose, D_o, is given by:

$$P(D_o) = \sum_{D_i \geq D_o} P(D_i) \tag{40}$$

The results of this computation for the two models described above appear in Figure I-7-28. From Figure I-7-28 it may be seen that the probability of exceeding a dose of 25 REM is 12.5 percent in the peak activity year. In the year of moderate activity this is reduced to 6.5 percent. This represents an estimate of the situation wherein all contributing factors are taken at their most conservative values. The only mitigating assumption is that of the two-day prediction capability. The present estimate of the probability of exceeding 25 REM for a year of peak activity is high. It may be seen however, that reduction in risk will be forthcoming from less cautious

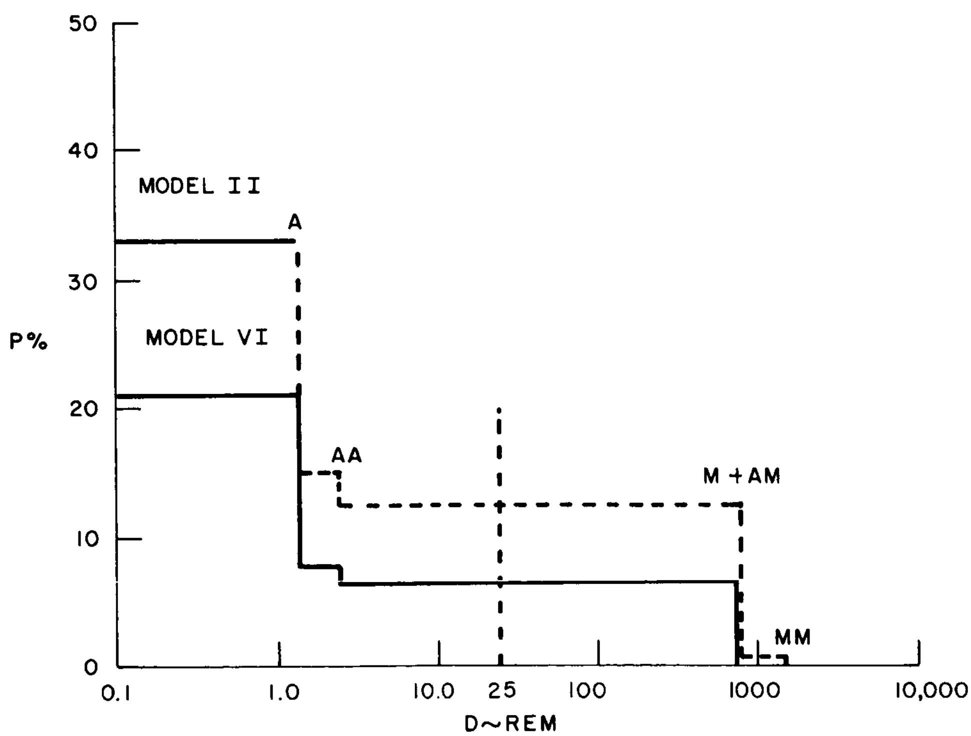


Figure I-7-28. Probable dose P = probability of receiving dose greater than D estimates of certain factors such as vehicle shielding and solar activity during the period of interest. This relationship may be taken as one constraint placed upon the scheduling of missions during the operational phases of the APOLLO program.

7.6 BIOLOGICAL IMPLICATIONS OF IONIZING RADIATION

7.6.1 Introduction

In this section, certain aspects of the biological problem are reviewed, particularly with a view toward measures to be taken in the event of radiation sickness in the crew. Although every effort must be made to solve the radiation problem by the means described above, biological alternatives are available and should be examined.

The term roentgen (r) has been used internationally for over 30 years. This unit is defined in terms of the ionization of air (1 esu/cc of air). One r is equivalent to an absorbed dose of 93 ergs/gm of tissue. Thus, the term rep (roentgen equivalent physical) was proposed as a unit to represent the absorbed dose physically equivalent to 1 r of x-irradiation. This term has been supplanted now by the term rad (radiation absorbed dose), which is 100 ergs deposited per gm. (US National Bureau of Standards Handbook, 62, 1956).

The amount of energy absorbed is not a good criterion of biological effect. The density of ionization along the path of a charged particle is also significant. This density is usually expressed in terms of the rate of linear energy transfer (LET) along the path of ionization (KEV/μ of tissue). The methods currently used for assessing biological effects of radiation involve measurement of the dose in rads and simultaneously exposing organisms or parts of organisms. Differences observed in the effects, where the rad is the same, may be explained as a difference in the relative biological effectiveness (RBE) of the radiation. One rad of 250 KV x-rays is considered to have an RBE of one. RBE is roughly equivalent to LET, but this relationship is non-linear.

Very little information concerning the RBE of space radiation (Van Allen and solar flares) is available. However, it is probable that the RBE for electrons is not greater than one. Attempts have been made to estimate the RBE of protons (see Figure I-7-29).

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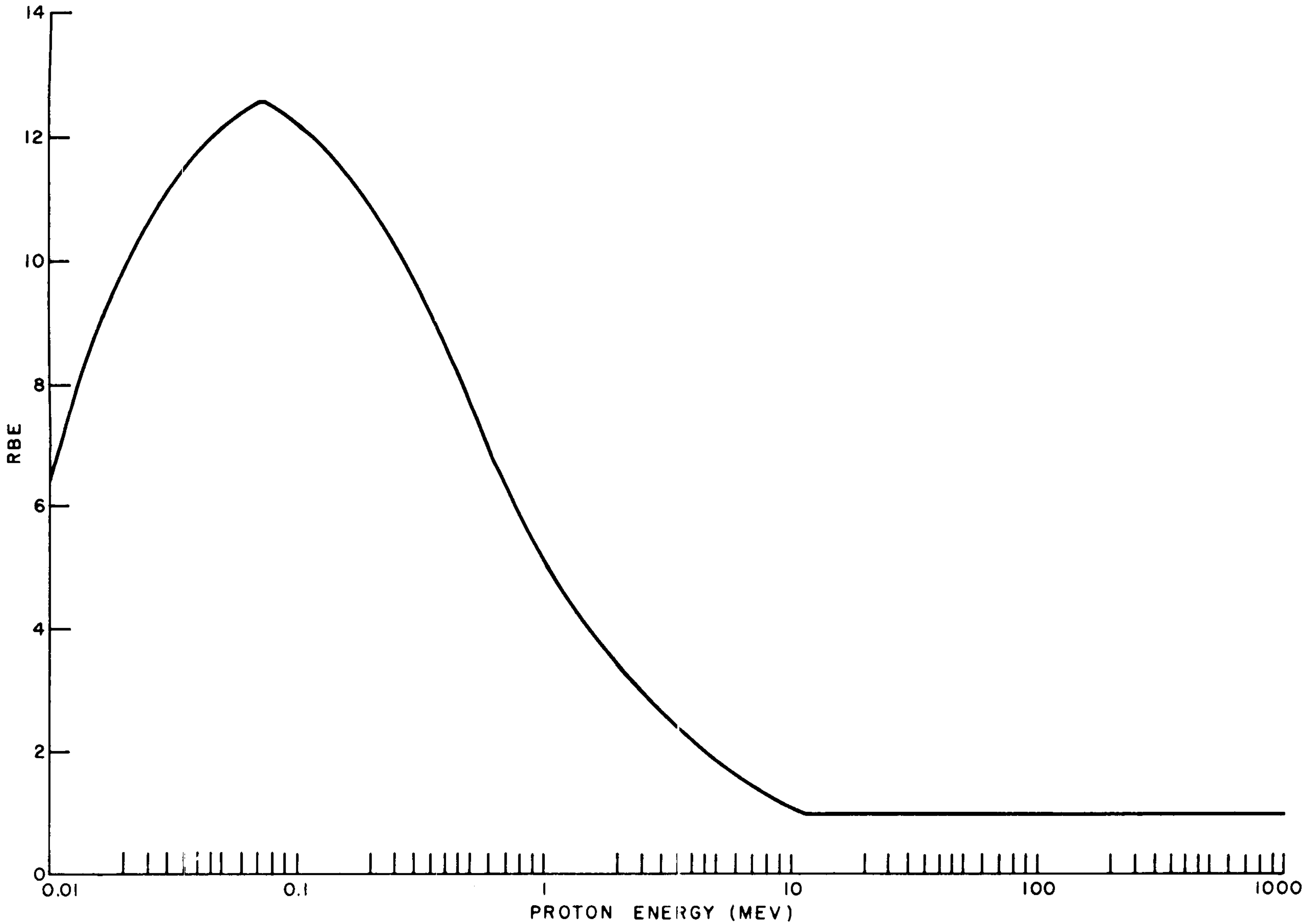

Figure I-7-29. Approximate relative biological effectiveness (RBE) for protons in soft tissue (After Rochlin and Goodale)

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Since the end of World War II, a vast amount of literature concerning the biological effects of ionizing radiation has been accumulated. The great majority of this literature describes the effects of x-irradiation and only a small portion is concerned with particulate irradiation. It is probable, however, that effects for all types of ionizing radiations are similar.

As a general rule, radiosensitivity of tissues is a direct function of the amount of cell division. Thus, gonadal tissue, hematopoietic tissue, and intestinal muscosa are examples of radiosensitive tissues. Brain and muscle may be considered as radioresistant. Three main types of lethal effects occur: (1) hematopoietic death, (2) intestinal death, and (3) brain or CNS death (see Figure I-7-30).

Sublethal effects, which are generally irreversible, include cataract formation, genetic damage and tumour production. Present standards of allowable levels of radiation are based on these and other sublethal effects.

7.6.2 Dose Limitations

The recommendations of the National Committee on Radiation Protection and Measurements are given below:

Exposure	Dose
(1) Critical organs (including whole body, head and trunk, active blood-forming organs, and gonads)	(1) 5 rems (age 18) 3 rems/13 weeks 5 rems/year 60 rems (18-30 years) 110 rems (18-40 years) 260 rems (18-70 years)
(2) Skin (whole body) and lenses of the eyes.	(2) 10 rems (age 18) 6 rems/13 weeks
(3) Extremities (skin dose)	(3) 75 rems/year 25 rems/13 weeks

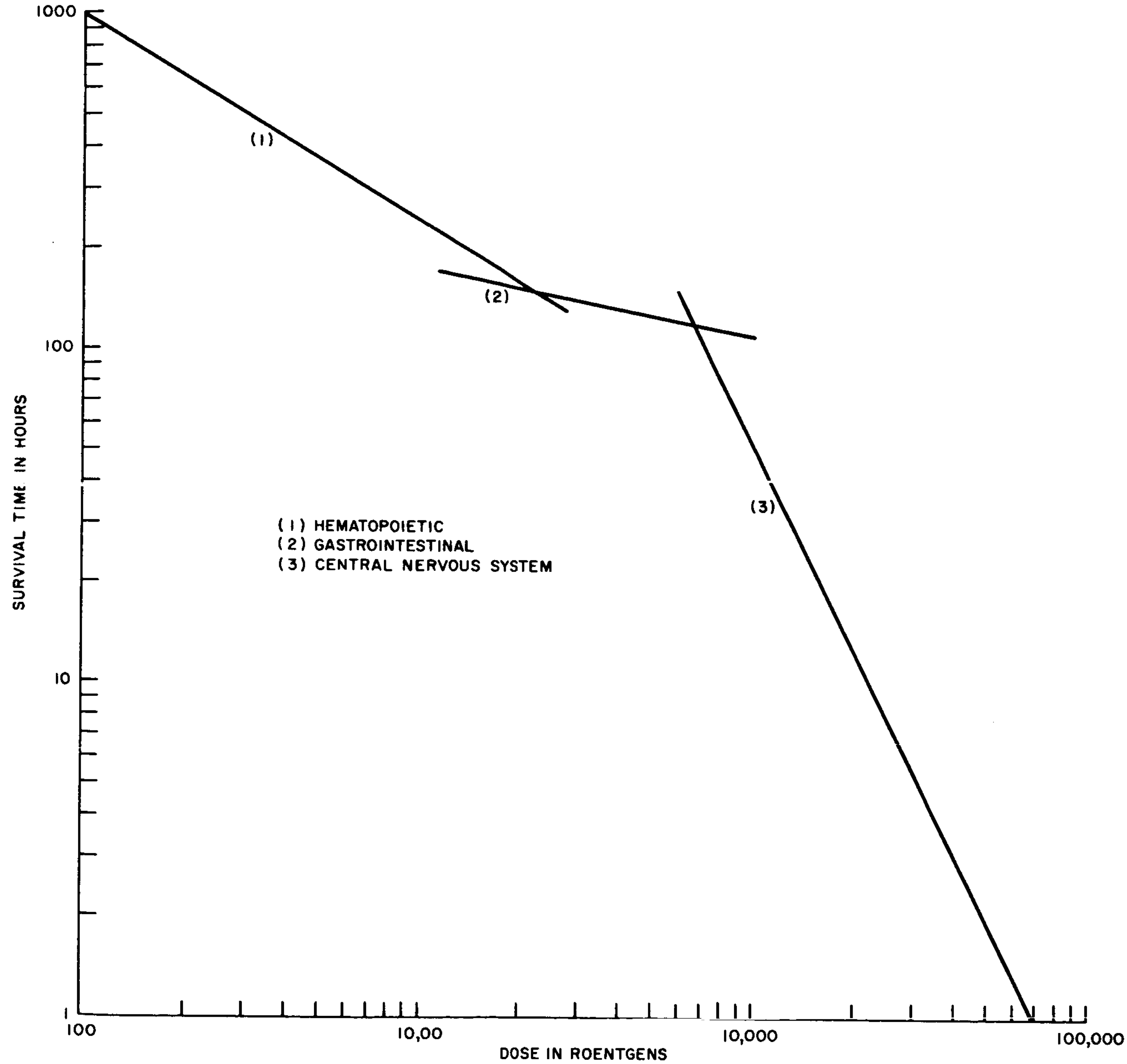

Figure I-7-30. Mean survival time after irradiation (After Pickering et al)

The emergency whole body dose is usually taken as 25 rems. Radiation dose levels in the table represent present occupational exposure limits which have been selected to produce no observable biological effects over a lifetime of work. Shielding considerations in space vehicles indicate that attempts to stay within the present pro rata occupational five rems per year may be extremely difficult (Tobias and Wallace, 1960). Radiation exposures of test pilots currently may result in a statistical shortening of life span of as much as twelve years. Such occupations as test piloting must have their own radiation standards.

Consequently, it has been suggested (Langham, 1960) that the benefits of space flight warrant increased risks and that standards of permissible dose be set more realistically. If crew efficiency is set as a criterion, a dose rate of 10 rem/day for a year will cause a few deaths, while a dose of 3 rem/day over a similar period will probably cause no deaths and no noticeable drop in operating efficiency. If higher radiation doses are to be considered, prophylactic and therapeutic measures for dealing with radiation sickness should be considered.

7.6.3 Chemical and Biological Methods of Protection

7.6.3.1 PROPHYLACTIC

The first definitive report of protection against ionizing radiation by chemicals given before irradiation was that of Patt and co-workers utilizing cysteine. Since that time, the literature has been flooded with reports of protection in numerous species with various chemicals. Much of this information has been reviewed by Patt and Maisin and Doherty. Among the compounds tested are thiols and disulfides related to cysteine and cysteamine, and pharmacological agents such as cyanide, and p-aminopropriophenone, histamine, 5-hydroxytryptamine and various metabolites. Of these, the cysteine-cysteamine group have received the greatest attention. The protective action of this group is fairly general, giving the appearance of a radiation dose reduction.

At the present time the isothiouronium derivatives, which undergo rearrangement to mercaptoalkyl-guanidines, appear to be the best protective agents. The two most prominent are 2-aminoethyl-isothiouronium (AET) and 2-mercaptoethylamine (MEA). As an example, Maisin and Doherty report that the LD 50/30 of mice is approximately doubled by use of 2-mercaptoethylguanidine (MEG) (152 mg/kg). Unfortunately, for the purposes considered the toxicity of these compounds is rather high. Condit and co-workers have demonstrated that humans given more than 500 mg of AET orally developed nausea and vomiting. Subjects who received doses of 750 to 1000 mg incurred a slight decrease in blood pressure. Dizziness, drowsiness, flushing, sweating, dyspnea, and diarrhea were also reported. If one may extrapolate from the mouse to man, a dose of about 12 gm (80 Kg man) would be necessary to double the LD 50/30 dose. Obviously, such a dose cannot be given to man if doses of less than one gram produce severe toxic effects.

7.6.3.2 THERAPEUTIC

In view of the fact that doses of about 1000 rem or less cause death by damage to the hematopoietic system (see Figure I-7-30) it appears reasonable to assume that injection of components of this system should be of therapeutic value up to this level. Irradiated animals lacking viable bone marrow actually die from infection, hemorrhage, anemia, or some combination of these disorders. Infection may be attributed mainly to the decreased number of granulocytes, hemorrhage, due to the loss of platelets, and anemia due to the decrease in red blood cells. Furthermore, there is evidence that new blood cell elements are formed from injected bone marrow in irradiated animals.

A circumstance that permits this type of treatment is that the primary immune response is injured to the extent that homologous and heterologous blood-forming cells can transplant (Makinodan, 1957; Wilson et al, 1959) and keep lethally irradiated animals alive. Long-term survival, however, is better when donor and host are isologously related. In some experiments (Congdon and Lorenz, 1959; Congdon and Urso, 1957) the animals treated with foreign bone marrow survive the acute radiation syndrome because of the transplanted marrow but develop a secondary disease process that kills many of the animals three weeks to three months after exposure. This pathogenesis is believed to involve an in vivo antigen-antibody reaction (Congdon, 1957). Furthermore, if an animal has received a midlethal irradiation dose, the injured immune mechanism sometimes recovers quickly enough to destroy homologous and heterologous bone-marrow cells (rat) after a temporary take. Actually, more mice were killed by the immune reaction than by radiation. Another limitation of this method of therapy is that bone-marrow therapy is ineffective against high levels of radiation which cause intestinal damage (Van Bekkum and Vos, 1957). The best that can be expected then, is to approximately double the LD 50/30 dose.

The time of injection of bone-marrow may be delayed several days, depending on the onset of bacteremia. In mice the bacterial invasion is most marked at the 2nd to 5th postirradiation day and continues up to the 15th day (Philipson and Laurell, 1958). Antibiotics are useful in delaying this invasion, but probabily do not decrease mortailty except in the sublethal range.

Other therapeutic measures involving non-regenerative tissues, i.e. transfusion are good supportive measures for the specific causes of morbidity in the radiation syndrome. Some reserach has been done with non-cellular compounds such as polyvinylpyrolidine (PVP), thiamine derivatives, and protein-free calf-spleen extracts but the results show only slight or no protection.

In summary, it may be stated that autologous or isologous bone marrow extracts given in conjunction with antibiotics (and perhaps blood cell infusions) should approximately double man's LD 50/30. If treatment is delayed, antibiotics should prove effective in delaying bacteremia up to about two weeks, depending on the level of radiation received. It should be pointed out that long-term effects such as cataract formation, nephrosclerosis, or tumour formation are apparently not affected by bone marrow therapy. Bone marrow therapy in humans involved in reactor accidents has been encouraging.

Another possibility which warrants further research is that of combined treatment, i.e. prophylactic and therapeutic. Burnett and Doherty reported that mice given AET before and bone marrow and streptomycin after irradiation survived 2600 r.

7.6.4 Personal Body Shielding

It has been demonstrated that shielding encompassing the whole vehicle may be excessively heavy if it is to be adequate for intense solar flares. Consequently, partial shielding of space vehicle occupants may be considered as a weight-saving device. Jacobson has demonstrated that shielding of the spleen in mice resulted in a substantial reduction of lethality. Shielding of the liver, head, or hind leg produced some decrease in mortality. It is reasonable to assume that shielding of the hematopoietic tissue should protect up to about 1000 rem. This is comparable to the protection afforded by bone marrow infusions. For protection against higher levels of radiation, the intestinal tract must be shielded. Albert and Swift et al have shown, for mice and rats respectively, that intestinal protection reduced mortality. On the other hand, Farr and DeBruin reported that protection of rabbit colon and small intestine had little or no effect on mortality.

A vest type of design may be necessary to protect the majority of the hematopietic areas within the body (this also protects the intestine). Such a vest is about 0.5 M^2 and its weight may be calculated. Foelsche has estimated that the dose in rep behind $15 g/cm^2$ for the 10 May 1959 solar flare would be 18 rep for the duration of the flare. If the vehicular shielding is $5 g/cm^2$, a $10 g/cm^2$ vest would be required. Such a vest would weighe 50 Kg or 110 pounds. For this case only the low energy flux is considered. If the high energy flux is considered, the dose is increased to 180 rep. Protection against such a dose requires approximately $31 g/cm^2$ shielding, or a vest of $25 g/cm^2$ having a weight of 130 Kg or 286 pounds. Furthermore, radiation may enter the shielded thoracic hematopoietic areas through the head, neck, and pelvic regions. The head is left unprotected as are the arms and legs. Although vehicular shielding may be increased and vest weight decreased thereby, vests for three men require considerable storage space.

7.6.5 CONCLUSIONS

1. Only vehicular shielding should be used.
2. Pre-radiation drugs have no proven value at the present time because of toxic effects of appropriate doses in humans.

3. Hematopoietic tissue transfusions should be of value up to about 1000 rem. For low levels (less than the LD 50/30) only autologous infusions should be made. Homologous transplantations may be used in the LD 50 to LD 100 range.

4. Transfusions (whole blood and plasma) and antibiotics are effective as post-irradiation supportive measures.

CHAPTER II

MAN-MACHINE INTEGRATION

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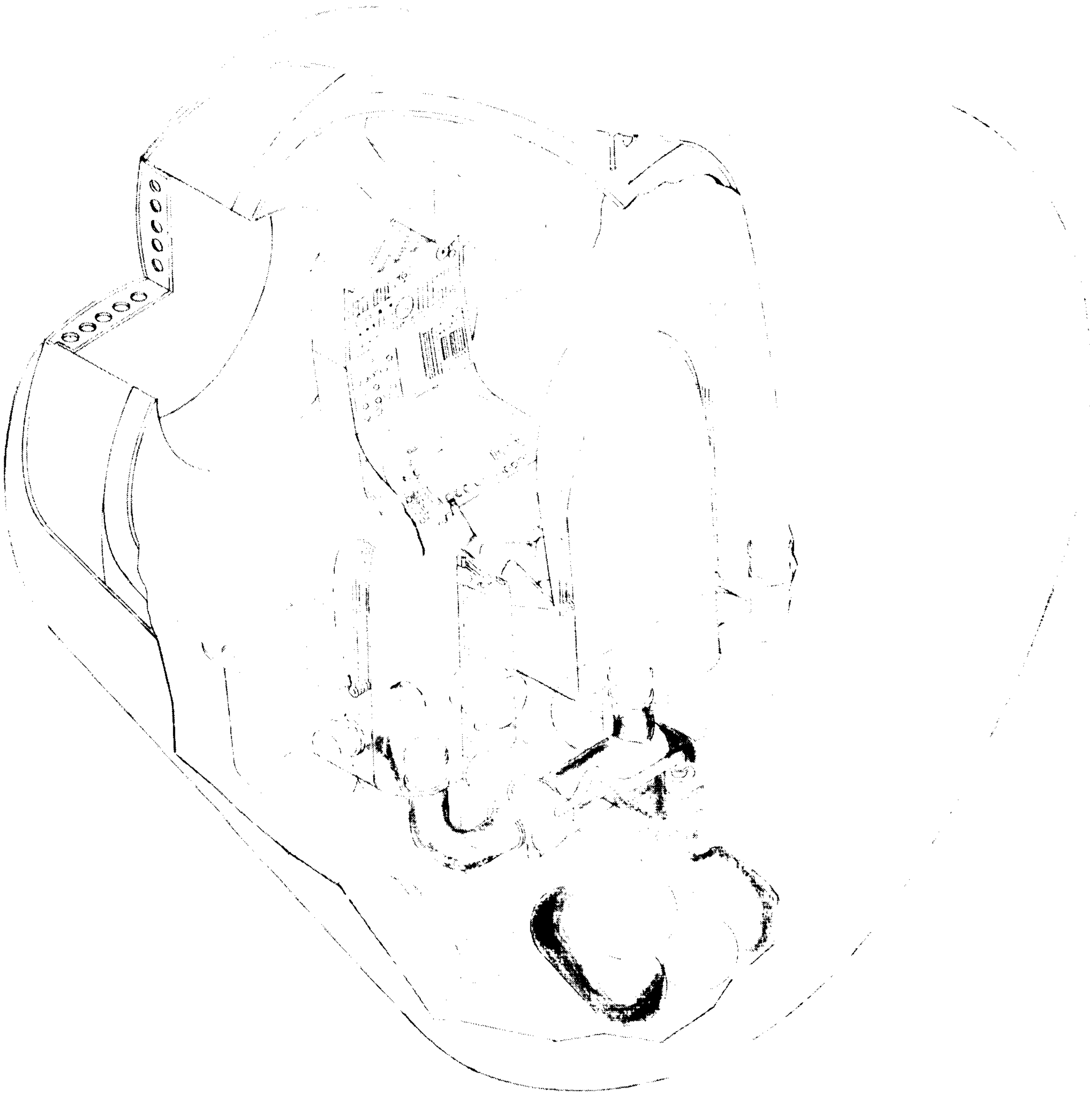

Cutaway view of re-entry vehicle showing instrument panel and controls arrangement

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II. PSYCHOLOGICAL CONSIDERATIONS AND MAN-MACHINE INTEGRATION

1.0 Introduction

The Engineering Psychology Program for the GE APOLLO study has been concerned with two basic approaches to the vehicle design problem. Initial efforts concentrated on a delineation of the problems of manned space flight as they are expected to occur aboard the APOLLO vehicle. This work considered the particular constraints and environmental conditions which can be expected to influence human performance.

On the basis of such investigations, and in conjunction with experience gained in a number of other Manned-Vehicle study programs (SMART, MTSS, GSS SLOMAR, etc.) the problems of man-machine integration were then studied.

The foundation of the study has been an analytical, sequential, and iterative procedure based on the methodology for control-display systems development which is promulgated by WADD; and which has been employed by Lear Inc. in their studies of a manned orbital bomber. The development procedure chart which outlines this method is presented in Figure II-1-1.

It should be clear, that while the depth and detail of the study have necessarily been less than that which must be conducted during the hardware research and development program, its scope has been broad. This technique not only provides the basic analysis for the control/displays system design as indicated in Figure II-1-1, but it also provides the analytical background for those phases of the study concerned with selection, training, maintenance, and habitibility.

Thus, the initial phases of the Development Procedure, e.g., those referring to mission profile, system requirements, and vehicle definition, have been broadened to include the examination of performance related factors relevant to manned space flight, (weightlessness, confinement, crew interaction, work-rest cycles, etc.)

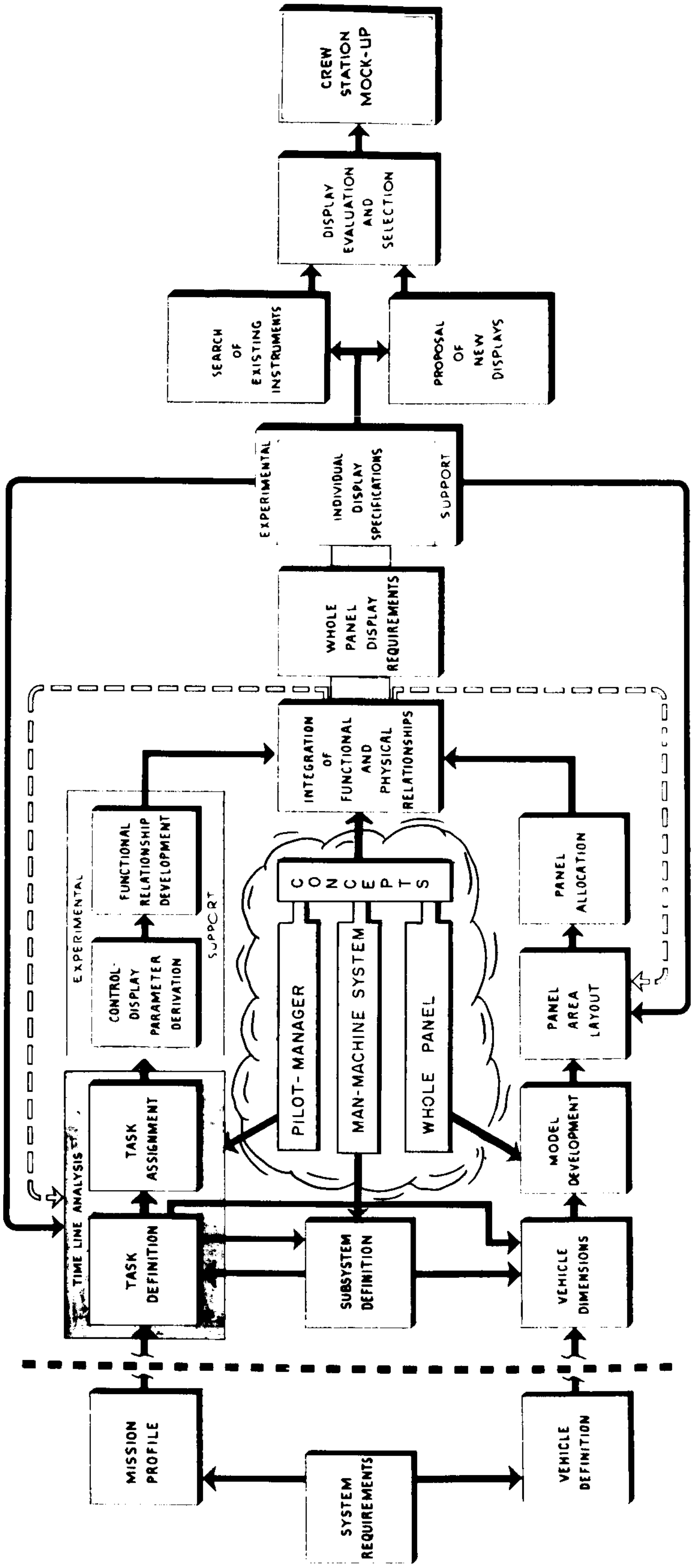

Figure II-1-1. Development procedures chart

The broad scope of the investigations conducted is represented by Figure II-1-2 which depicts the overall nature of the Engineering Psychology program. The present study includes coverage of every aspect of the program depicted in this figure. Primary emphasis however, has of necessity been given to selected areas of investigation as can be seen from examination of the table of contents, and the reading of this report.

Based on design guides, preliminary knowledge of the subsystems, and consideration of the nature of conditions which would influence performance, an analysis was made of those tasks which man might be expected to perform during each phase of the mission. The analysis was essentially a stepwise procedure which began with the mission profile. It specified potential limitations, defined preliminary accuracy requirements, and stated which subsystems might be involved. As the vehicle became better defined, only those subsystems which were retained in the vehicle were considered in the analysis. Subsequent analysis was conducted on the basis of the subsystems, where manual inputs to these systems were noted, and the involvement of other subsystems was indicated. Further analysis involved specification of the time required to make manual inputs, and the information upon which these inputs would be based. Finally, the possible sources of display information required for each input were stated.

A sample of this analytical work describing the launch phase of the mission is included in Appendix HF-I of this volume. All phases of the mission were investigated in this manner. In addition, for each specific subsystem under consideration charts were established to indicate the nature of the manual inputs, the information required to effect such inputs, and the possible sources of the information.

A preliminary time line analysis was also conducted and included in the Midterm Report. An updated form of this analysis, Table II-1-I, in somewhat greater detail, is included to indicate the general nature of the crew functions throughout the course of the mission. The role of the crew in the vehicle system will also be apparent from examination of the modes of operation for the display and control system, the approach taken to the maintenance problem, and the outline of the selection and training program.

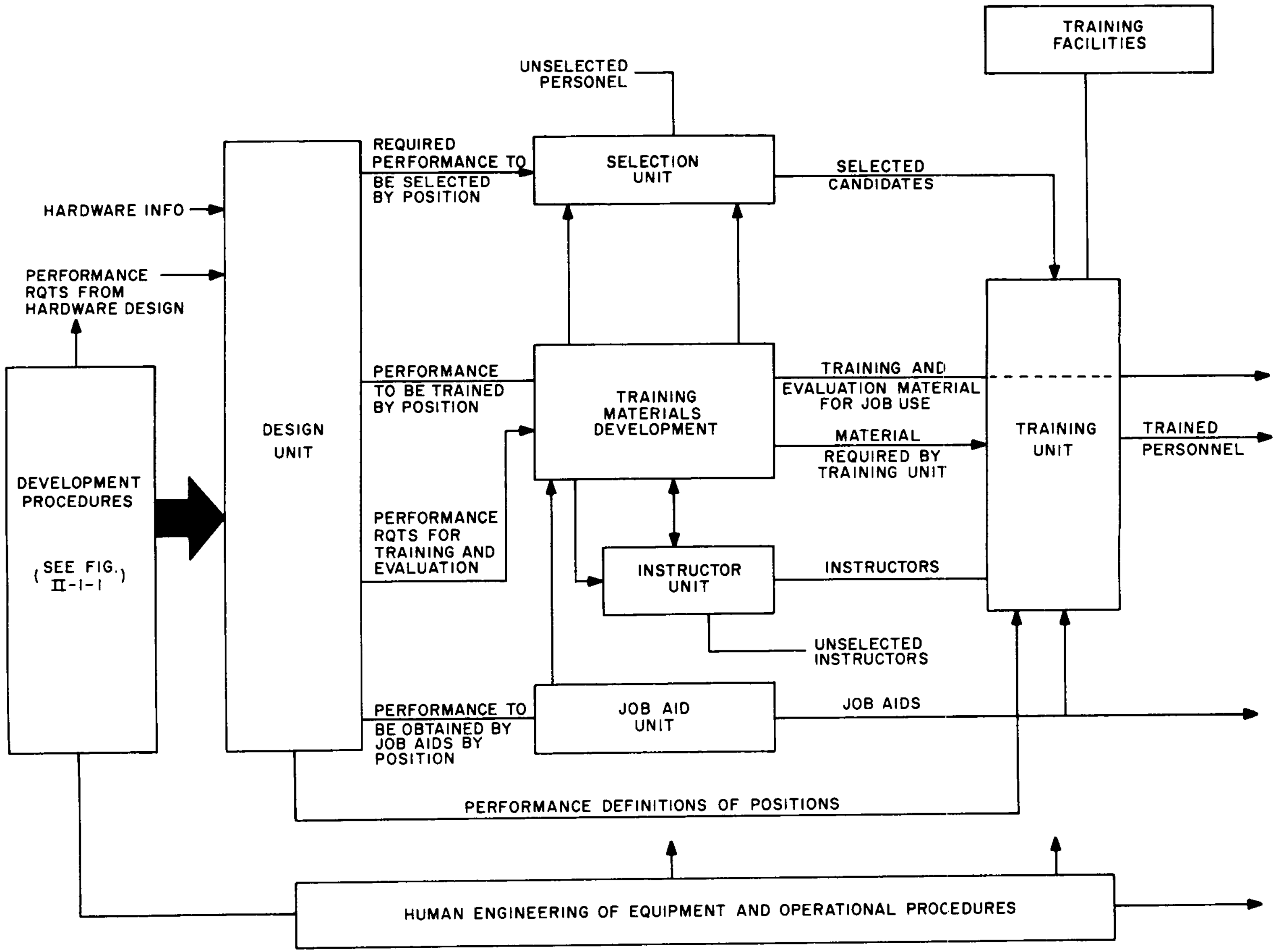

Figure II-1-2. Engineering psychology study program

[redacted]

TABLE II-1. UPDATED TIME LINE ANALYSIS

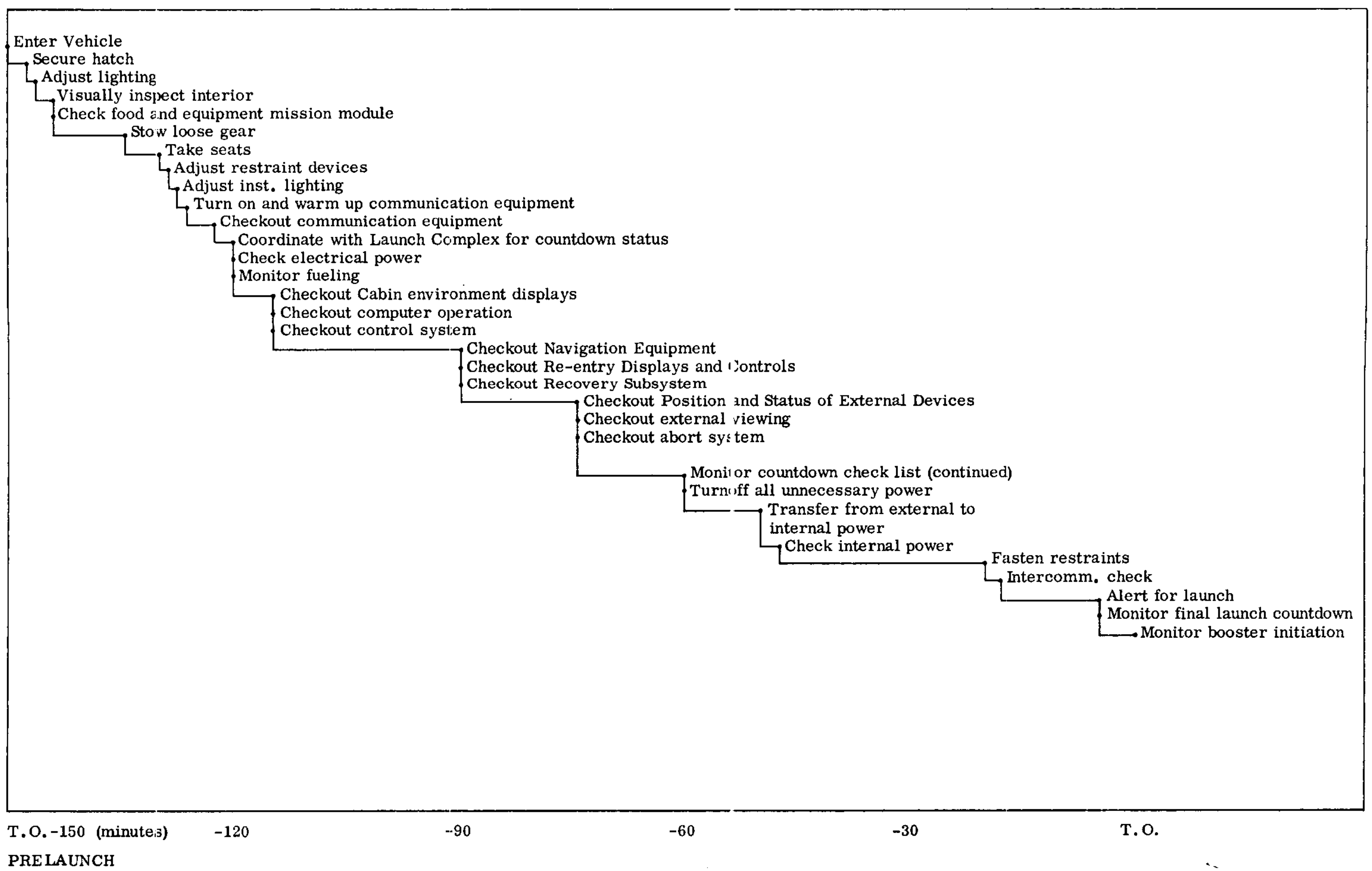

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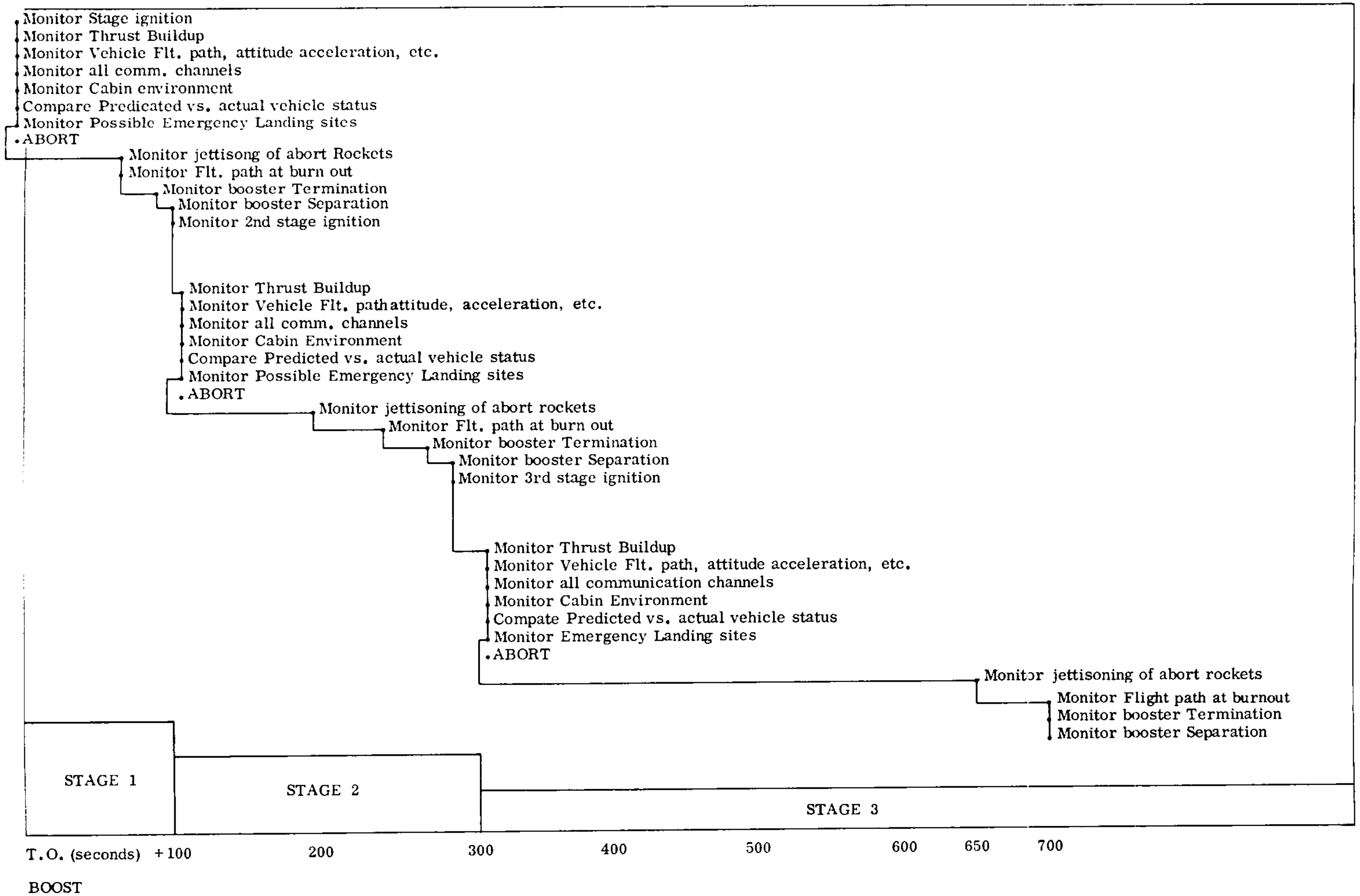
TABLE II-1-I (Continued)



[redacted]

TABLE II-1-I (Continued)

- Orient Vehicle attitude
- Monitor Vehicle Flt. path
  - Extend antenna and peak tune
  - Extend solar collector
    - Inflate solar collector
    - Loosen restraints
      - Inspect Vehicle interior
        - Check communication subsystem
          - UHF
          - VHF
          - TV
          - Telemetry
          - intercom
        - Check Rocket power subsystem
          - Rocket engine (actual vs. Programmed)
            1. pressure
            2. condition
            3. gimbal position
          - Check Fuel and Oxidizer System (actual vs Programmed)
            1. Flow rate
            2. Total consumed
            3. Total remaining
            4. Time to Pressurize
            5. Fuel - oxidizer pressure
          - Check secondary power
            1. Monitor input from solar modules
            2. Capacity-fuel cell batteri‹s
               - hydrogen pressure
               - oxygen pressure
            3. Output of fuel cell batteri‹s-solar modules
               - voltage
               - current
            4. Circuit breakers
            - Check environmental subsystem
              1. Lightir g
              2. Air conditioning
                 - gas supply (primary) flow rate & quantity
                 - gas supply (emergency) flow rate & quantity
                 - Temperature (atmospheric and wall temperature)
              3. Cabin pressure
                 - monitor - regulate total pressure
                 - pO_2
                 - pH_2
                 - pCO_2
                 - Leak detection
              4. Toxic gas
                 - Sensors
                 - Removal system
              5. Fire detection & control system
              6. Radiation and ion balance

T.O. 13+00 (hours) — 14+00 — 15+00 — 16+00 — 17+00

MIDCOURSE (60 hrs. ± 5)

[redacted]

TABLE II-1-I (Continued)

Check out and align astro-tracker
Orient vehicle for astro-tracker fix
Smooth, correct and plot astro-tracker fix
Prepare for manual check of position
Make celestial observations
Plot celestial observations
Determine need for correction
Monitor predicted vs. actual position
Determine time for correction
Crew alerted for course - change, in seats, and restrained
Prepare for velocity correction (check list)

1. Select VELOCITY CORRECTION on sequence display
2. Check power distributed panel observing buss load displays
3. Select COMPUTER or MANUAL input thrust control system if MANUAL
   a. Select DATA LINK on computer readout or obtain parameters by voice communications
   b. Set magnitude
   c. Set azimuth angle
   d. Set elevation angle
   e. Set time to fire
   f. Select ROCKET FIRE on time-to-go mode selector
4. Check thrust level on engine desired
   a. Select "THRUST CHAMBER" mode on commanders side stick
   b. Observe chamber position on display
   c. Manipulate stick to obtain normal pominal position
5. Select THRUST CONTROL on command needle mode selector
6. Select THRUST CONTROL on flight control mode selector
7. Observe command needles, attitude display, vehicle rate display
8. Pressurize desired fuel and oxidizer tanks
9. Open fuel valves from pressurized tanks
10. Select "LUNAR AIM POINT" on computer readout
11. When time-to-go approaches zero, turn on ignition system
12. At 5 seconds to go, actuate normally open safety switch on commander's left armrest

As rocket fires, observe;
a. Thrust
b. Total velocity on computer readout
c. Command needles on attitude indicator
d. Body axis rate display

Release safety switch, if automatic cut off fails, or if for any reason premature cut off is desired
Extend solar collector
Select SOLAR CELL or command needles
Select SOLAR CELL on flight control system
Check vehicle re-orientation
Check solar cell out put on signal strength meter
Check charge rate of individual fuel cell batteries
Select antenna on commanders side stick
Observe antenna position on signal strength display
Rotate antenna about one axis to achieve maximum signal strength
Adjust gain of Signal Strength display as desired
Rotate antenna about other axis to achieve maximum signal strength
Repeat about first axis
Inform ground station of all course changes
Release restraints
Resume normal operating mode - Eat, personal hygiene, maintenance as required, etc.
TIME UNDETERMINED

USE PROCEDURES OUTLINED ABOVE FOR ALL MIDCOURSE CHANGES EXCEPT FOR FINAL MIDCOURSE CONNECTION PRIOR TO LUNAR ORBIT.

T.O. + 1900 (hours) T.O. + 1800 | 19+00 | 20+00 | 21+00 | 22+00 | 23+00

MIDCOURSE (Continued)

[redacted]

TABLE II-1-I (Continued)

Orient vehicle for astro-tracker fix
Smooth, correct and plot astro-tracker fix
Prepare for manual check of position (sextant)
Make celestial observations
Plot celestial observations
Monitor predicted vs. actual
Check position velocity
Checkout functioning of all lunar sensing and experimental equipment
Calculate moon orbit parameters
Select LUNAR ORBIT on sequence display
Select LUNAR ORBIT on computer readout
Check power distribution panel
Check secondary power system
Evaluate fuel requirements to achieve desired orbit
Crew in seats and restrained
Initiate velocity change to achieve orbit (follow check list described previously)
Check program for lunar circular orbit

T.O. 57+30 58+00 59+00 60+00-5 hrs.

FINAL MIDCOURSE CORRECTION PRIOR TO LUNAR ORBIT

[redacted]

TABLE II-1-I (Continued)

- Check parameters of moon orbit
  - Establish lunar work schedule
    - Release restraints
      - Assume lunar orbit work stations

Lunar Lightside

- Orient antenna
  - Observe antenna positions and signal strength display
  - Rotate antenna about one axis to achieve maximum signal strength
  - Adjust gain of signal strength display as desired
  - Rotate antenna about other axis to achieve maximum signal strength
  - Repeat about first axis
- Orient solar collection
  - Check solar cell output on signal strength meter
  - Check change rate of individual full cell strength display
  - Check environmental display - adjust if required
- Adjust vehicle altitude so that moons surface can be observed and photographed (as required)
- Observe moon through periscope
- Verify altitude reading with periscope
- Begin lunar orbit observations and experimentation (not defined)
  - Initiate cameras, data recording devices TV etc.
- Maintain communications with earth stations, when possible, relay data
- Monitor operation of mission equipment
- Perform unscheduled maintenance as required

Lunar Darkside

- Check operation of fuel cell batteries when transiting between dark and light portion of orbit
  - Turn on tape recorder when behind moon
    - Check environmental displays - adjust if required
      - Begin lunar orbit observations and experimentation
      - Monitor operation of mission equipment
      - Perform unscheduled maintenance as required

Upon completion of final orbit ⟶

- Perform subsystem checkout in preparation for de-orbit
- Crew in seats and restrained
  - Select "EARTH AIM POINT" on computer readout
    - Select time for lunar orbit departure time
    - Initiate velocity change

MIDCOURSE CORRECTIONS - INBOUND FROM LUNAR ORBIT

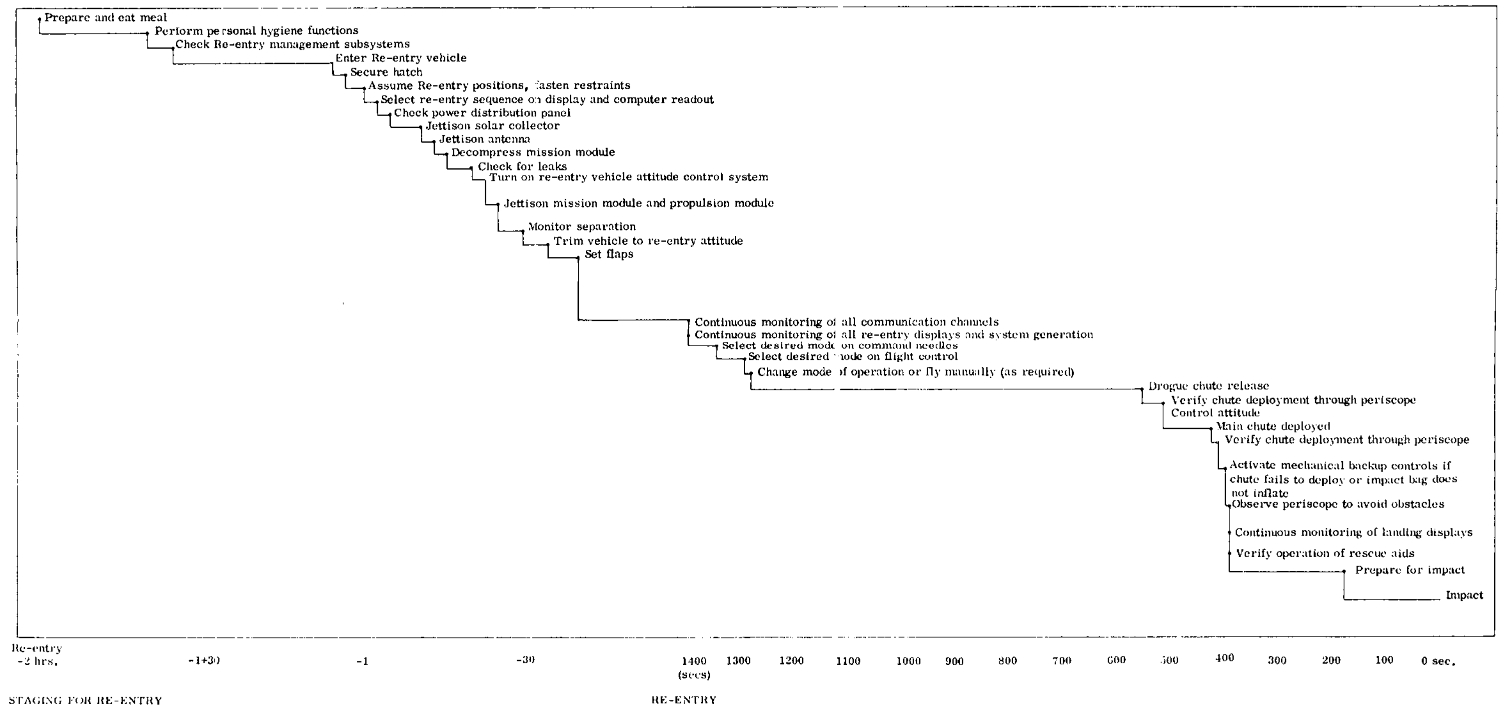
SIMILAR TO MIDCOURSE FLT - III

60±5 hrs.	61+00	62+00	63+00	64+00	hrs.
	1+00	2+00	3+00	4+00	

LUNAR ORBIT

[redacted]

TABLE II-1-I (Continued)



Initially, the philosophy of the use of the man in the system was general in its statement. This general statement was carried forward into a great many specific cases of detailed design; for the allocation of functions between man and machine is not in the final analysis a philosophical problem, but rather is a matter of hardware design. It is part of the purpose of the present studies to show what issues need to receive attention in the research phase of APOLLO in order that specific hardware design will indeed embody the defined philosophy of the role of the man in the system.

The consideration of man's role in the APOLLO system was begun by acknowledging that he will be in the vehicle. The definition of the mission presumes his presence in the vehicle, and this permits design of equipment to involve his participation in vehicle control anytime such participation will result in a lighter, more effective, more reliable vehicle. The design of the equipment for any control action incorporates the man when and only when the overall criteria are better served. The detailed design will always involve a comparison of the way the control function could be accomplished without the man. Some control functions must clearly be assigned to the equipment, for example, when man is unable to respond quickly enough or with the required degree of accuracy. Some other control functions may clearly be assigned to the man, for example, when the necessary hardware to perform this function imposes inordinate weight or power penalties. This will leave a great many control functions which may or may not be assigned to the man, and the decisions as to the distribution of these functions are part of the detailed design for APOLLO equipment.

Out of past experience some rules of thumb have been developed regarding allocation of functions between man and machine. In general, it may be stated that where the criteria for a control action are very clearly defined and are present in the situation, and where the control action is to be accomplished the same way every time, then this control function should be performed by the equipment. On the other hand, when the conditions for a particular control are not clearly defined, may change from time to time, or may involve some criteria not present in the situation, or where there is more than one acceptable way for the control action to be carried out (especially when the preferred control mode may change from time to time), then the control action should include the human operator. At this stage of analysis, there is still not enough information to go directly into detailed design. The decision to use a man in the system interacts with decisions about what kinds of information should be displayed and what kind of control action should be used in the system. Most of these decisions have been made, and they and their rationale are given in this report.

In the APOLLO study, consideration has been given to the use of man in each of the following subsystems:

1. Initial boost
2. Throttleable boost
3. Descent parachute
4. Boost deflection attitude control
5. Unpowered trajectory attitude control
6. Aerodynamic attitude control
7. Drag deflection
8. Parachute deflection
9. Optical position fixer
10. Optical attitude and altitude sensing
11. Optical sensing of moon surface and other mission objectives
12. Communications
13. Data transmission
14. Launching the moon surface probe
15. Camera recording
16. Sensing equipment malfunction
17. Sensing re-entry heating
18. Sensing quantities of required stores
19. Biomedical sensing
20. Sensing characteristics of the atmospheric environment on re-entering

[redacted]

21. Operating the inertial position computer

22. Operating the orbital elements computer

23. Selecting the status of automatic attitude stabilization

24. Programming thrust cut-off

25. Programming the initiation of controllable thrust

26. Operating the energy management computer and

27. Life Support System

Perhaps the most adequate summary statement which can be made concerning man's overall role in the vehicle system is that in general he performs command functions rather than carrying out control functions. Thus he is provided with information which permits him to monitor vehicle status, to make appropriate decisions, and to implement the appropriate course of action but not necessarily to carry it through as a control function. Stated differently, we have utilized the "pilot-manager" concept in our approach to crew function.

The general design philosophy, in providing practicality, flexibility, and reliability to the system through employment of the human operators, establishes an essentially active role throughout the mission for each operator in the crew.

The man-machine interface described in this volume applies equally to the D-2 vehicle or the R-3 vehicle in terms of method, approach, psychological considerations, and almost all aspects of control and display design. There are some exceptions. The allocation of functional display-control functions to the physical panel areas differs in the R-3 and D-2 because of the differences in vehicle structure. In addition, landing a glide re-entry vehicle, (R-3), requires instruments which are not necessary in the semi-ballistic, (D-2), vehicle. This requirement is discussed in the body of the report. The differences in the landing portion of the mission for the two vehicles also implies differential training of the crew for control during this mission phase.

Aside from these considerations, the functional and operational requirements are essentially similar for both vehicles. Thus the results presented in the body of this report, with the exceptions noted can be considered applicable to both vehicles.

[redacted]

2.0 CONTROL-DISPLAY SUBSYSTEM DESIGN

2.1 PROGRAM DEFINITION

2.1.1 Introduction

The purpose of this section is to describe the development procedures and the control-display subsystem which has been evolved in the course of this study. It serves to define the man-machine interface for the General Electric Co. APOLLO vehicle as developed by the Advance Engineering Division of Lear, Inc. working under the direction of the General Electric Co. Human Factors personnel. The controls and displays specifically required for a manned lunar orbit mission were given prime consideration in this effort, while secondary consideration was given to the additional requirements for lunar landing or earth orbital missions.

In order to describe the development procedure and the resulting control-display subsystem, this section is divided into five sub-sections. The first sub-section is devoted to a discussion of the ground rules established for control-display subsystem development of the APOLLO vehicle while the second sub-section consists of a general description of the application of the "Mark IV" design method to the APOLLO system. The analytic outputs of the design method, are included as sub-section 2.3 which describes in detail the physical and functional parameters of the display-control system. The resulting crew station and display panels are described and illustrated in the fourth sub-section of this report. Sub-Section five, outlines areas where further research is required.

2.1.2 Design Objectives and Ground Rules

A preliminary but important aspect of control-display subsystem design is the establishment of a set of ground rules. With respect to the APOLLO vehicle, these ground rules were obtained from various statements concerning system constraints and design philosophy.

Because they aid design engineers in arriving at design criteria and in making trade-offs between alternative sets of proposals and equipments, the control-display subsystem that is developed in this report reflects these ground rules.

2.1.2.1 FEASIBILITY

Dominating APOLLO control-display design is the concept of feasibility, and in particular, the feasibility of utilizing devices selected as the means of instrumenting suggested displays. In this respect, reliance on current state-of-the-art is not considered to be a matter of overriding importance. Advances in the state-of-the-art are recognized as being necessary in order to permit the design and selection of the most promising displays in terms of reliability and human factors considerations.

It cannot be assumed that instruments which are highly reliable under present flight conditions are also likely to be highly reliable under the vastly different conditions of space flight. It is probable that mechanization techniques which are in the early stages of development today may prove to be more reliable on future missions than instrumentation utilizing current state-of-the-art processes. This is because of inherent superiority in some of their design characteristics.

Another reason for stressing the concept of feasibility is that current state-of-the-art instrumentation may not be reliable enough. Although a known reliability coefficient can be established for some particular equipment, this equipment may still not be adequate within the system. This imposed requirment does not effectively remove from consideration instruments which are practical within the present state-of-the-art, but implies that all practical possibilities should be considered, including equipment whose reliability can only be estimated.

Perhaps the most important reason for emphasizing the concept of feasibility is that display design is not solely restricted to the use of current mechanization techniques. Primary attention must be given to the design of meaningful and readily interpretable displays which incorporate sound human factors principles in the development of integrated displays and display panels. Thus, new displays which evolve may be more complex than those which appear permissible if only current state-of-the-art techniques are assumed feasible.

2.1.2.2 BACKUP PROVISIONS

Events where component and subsystem failures occur cannot be ignored. To take into account possible component and subsystem failure, it appears desirable to build into the system adequate redundancy and back-up modes of operation. One of the principle considerations with respect to building into the system various back-up modes is the notion that redundancy should be designed into the optimum mode of operation. This may be contrasted to the desirability of establishing numerous degraded back-up modes in which the operator enters the degraded loop. In the attempt to establish overall high system reliability, there is a trade-off, of course, between designing redundant but identical system control loops (the optimum system) and providing the operator with the capability of entering into the loop at different places with the prerequisite supporting equipments.

The approach taken in the APOLLO control-display design was to consider the performance decrement that would result by not duplicating the optimum system, and the cost, in terms of weight, and volume, of duplication. If performance was degraded to the extent that continuance of the mission was not practical, duplication of the optimum system was utilized. If, however, means could be provided which permitted the operator to enter the degraded loop without a significant loss in performance, the optimum system was not duplicated and a back-up mode of operation was evolved.

2.1.2.3 POWER, WEIGHT AND VOLUME

Wherever possible, power, weight and volume limitations were explicitly taken into account in the design of the control-display subsystem. This was accomplished by a rating technique which utilized engineering estimates to rank order alternative means of instrumenting the various parameters required for display.

Micro-miniaturization might be employed in the design and production of more advanced instruments in order to meet these system constraints. However, the art of micro-miniaturization is in such a state of flux at this time that it precludes an accurate estimate of potential capabilities. For this reason, more conservative display techniques have been used with the notion that transition to new techniques would be more easily accomplished than if current estimates were heavily weighted with micro-miniaturization considerations.

In regards to advanced display techniques, there is one which appears particularly promising at present because it presents a potential savings in weight and volume without a corresponding loss in reliability. In addition, it seems highly applicable to the environments which will be encountered in space i.e., zero gravity and pressure (as a result of cabin decompression). Specifically, electroluminescent displays look promising and should be thoroughly investigated. In fact, it seems likely that if their present rate of development continues, by the time of the expected launch date, electroluminescent instrumentation will be adequate for the presentation of many of the parameters which are displayed in the APOLLO vehicle.

In some cases, control-display subsystem performance would be limited if more complex and integrated displays were not permitted. For this reason, more advanced instruments were utilized to display some of the complex relationships between critical display parameters. Nevertheless, the use of complex and highly integrated instruments was kept at a minimum.

2.1.2.4 MAINTAINABILITY

Maintainability of instruments has also been a design objective. It has been approached in control-display subsystem design by providing, through design and layout, for access to all black boxes and display panel areas, and by utilizing a modular approach to instrument packaging. This last consideration makes it possible to minimize the number of spare components required in the event of failure. In total, the maintenance problem has been considered an integral aspect of panel and individual instrument design.

2.1.2.5 FLEXIBILITY

System flexibility was another design objective. Two considerations were taken into account. The first was to facilitate through control-display design the possible application to future APOLLO missions of major segments of the developed control-display subsystems. This does not imply that the representative man-machine interface developed for APOLLO is capable of a lunar landing mission. Rather, it indicates that many of the subsystem displays and controls which will be developed for the lunar orbit mission are also applicable to the other APOLLO missions.

Another consideration involved the design of the control-display subsystem so that alternative (secondary) missions could be achieved in the event that it was not feasible to continue on a lunar orbit mission. This is especially true in the case of abort trajectories which might be necessary during any mission. In this sense, control-display flexibility is required which permits numerous trajectories other than nominal. Controls and displays have been designed to maximize the inherent flexibility of the men in the system.

2.1.2.6 ENVIRONMENT

Instruments must be designed which will be able to withstand all applicable environmental tests including the special constraints imposed by zero "G" and zero pressures associated with the space environment. This requirement for the design and development of instruments able to withstand an extreme range of environments will influence final instrumentation for the APOLLO system a great deal. For instance, one of the most critical factors in design concerns the arcing problem encountered with high voltages in the zero pressure environment. Coupled with the already existing requirements associated with vibration and shock resistance, etc. the requirement for rugged hardware is obvious.

2.1.3 Time Schedule

Time, of course, is an important constraint in any system design and development program. Consequently, state-of-the-art projections may be optimistic or pessimistic, depending upon the point of view one wishes to follow. For the initial control-display subsystem design, as reflected in the mock-up of the crew station, it was considered advisable to provide the most feasible and practical approach to the problem. In this sense, estimated instrument development time and effort was kept at a minimum in the interest of designing an adequate system. Thus, only the most promising and practical of future display techniques were employed. It should be mentioned, however, that state-of-the-art projections were very optimistic with respect to the area of required sensors and computational techniques. Consequently, it was assumed that it would be possible to sense and/or compute all required parameters within the time schedule provided.

2.2 DESIGN PROCEDURE

The methodology applied throughout this phase of the control-display research and development was adapted from an established procedure which Lear Inc. helped to develop and design for the USAF.

Figure II-2-1 is a block diagram illustrating the design procedure that was used in developing the APOLLO control-display subsystem. The purpose of this sub-section is to describe how this design procedure was followed and to what extent each individual step in the method was completed. This procedure is inherently a re-iterative process and depends upon the systematic re-cycling of the steps for complete definition of the control-display subsystem. As only one cycle has been completed, the resulting APOLLO control-display mock-up described in Section 4.0 is not a final statement of this subsystem.

2.2.1 Panel Area Layout

System requirements have resulted in the definition of the APOLLO vehicle and description of various mission profiles. The vehicle was defined in terms of drawings describing its structural configuration and dimensions. From these drawings a preliminary but full scale model of the D-2 crew station was constructed which defined the space limitations of the crew station in the recovery vehicle. Figure II-2-2 shows a picture of the model.

Within the area defined by the model, three dimensional control and display panels were shaped and alternative panel areas and locations were examined. Panel areas were studied with respect to the factors of size, distance and angle from defined visual reference points, ease in gaining entrance to the mission module, accessability (front and back) of panel areas, ability of panel to be time-shared by several members of the

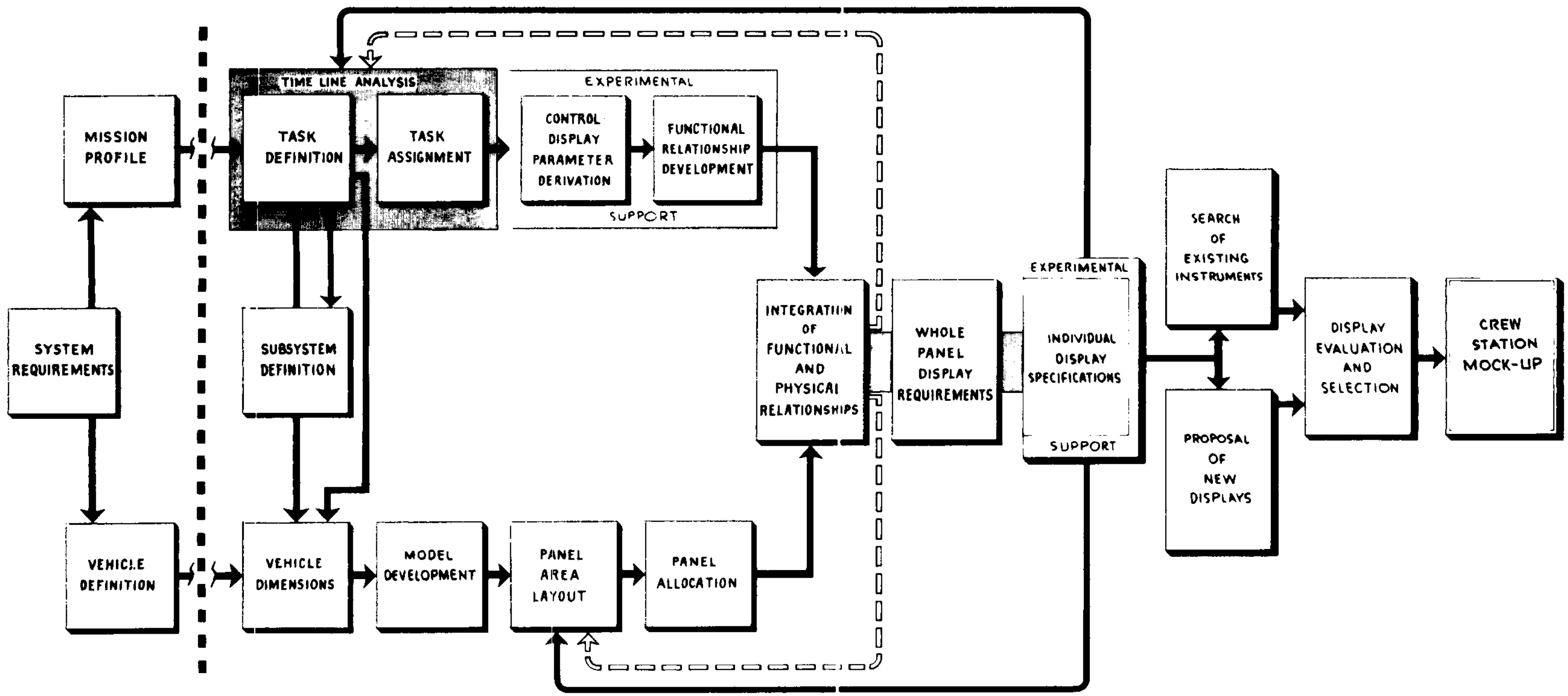


Figure II-2-1. Development procedure chart

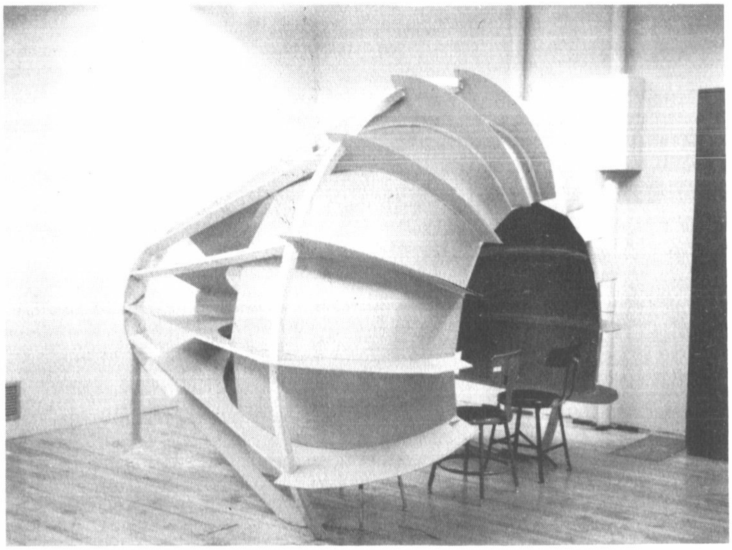


Figure II-2-2. Full scale R-V model

crew, adaptability to the secondary pressurization cocoon, and distribution of crew functions. The result of this process, Figure II-2-3 is a defined panel area which not only adequately meets the above requirements but is also aesthetically pleasing in appearance.

2.2.2 Information Requirements

The mission profile of the APOLLO system was defined in terms of trajectory analysis and mission objectives while subsystems were defined by personnel responsible for their development. From these inputs, functional allocation charts (Paragraph 2.3) for the various subsystems were derived. These charts define, for each mission phase, the general functions which must be performed by the particular subsystem, and areas where the crew is likely to interact with the subsystem in performing their control and monitoring tasks. This is the first step in arriving at a time line analysis to define and assign specific tasks to the crew members in order to evaluate loading effects upon the crew in the developmental system.

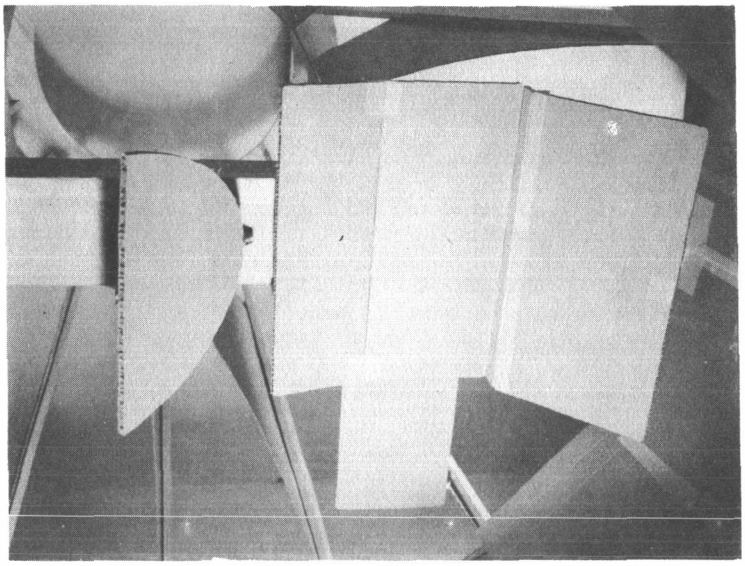


Figure II-2-3. Defined panel area

A detailed time line analysis was not performed at this stage of control-display subsystem development because specific displays and controls were not defined to the extent that would permit more than general operational statements. A more detailed time line analysis would be conducted, of course, during the next cycle of this design method where a crew station mock-up would be available. This would be used for obtaining an adequately detailed description of the crew tasks and for making the appropriate trade-offs in cases where work load upon the crew members is too great.

Control-display parameters required for the APOLLO system were derived from the functional allocation charts, and are listed in the Control-Display Parameter Specification Charts (Sub-section 2. 3). They were obtained by reducing the general functions described in the functional allocation charts into the logical dimensional characteristics which reflect the separate information and control requirements for each function.

The reduction of general and descriptive functions to individual control-display parameters offers the extra advantage of permitting the discovery of additional relationships between parameters which might otherwise go unnoticed. To increase this possibility, individual control-display parameters within a subsystem are cross-referenced to other control-display parameters in that as well as other subsystems.

The relationships which have thus far been established (cross-referenced) for the APOLLO system are given in the Control-Display Specification Charts, (Sub-section 2. 3).

2.2.3 Integration of Functional and Physical Relationships

Once panel areas are defined and functional relationships developed, there begins the task of integrating the required controls and displays within the available panel and console area. This is accomplished with the expectation of producing a crew station layout that meets both anthropometric and operational considerations. The design technique involved a process of locating various control and display parameters in different panel and console areas and then observing the results. Helpful in this process was the logic of designing panels for individual subsystems or other groups of functionally related parameters. The result of this process is reflected in the mock-up of the APOLLO control-display subsystem.

2.2.4 Individual Display Specifications

The approximate size of the display area for individual parameters can now be determined. In addition, where it was desirable, because of the degree of interdependence, parameters were integrated in designing meaningful displays. These factors, plus overall weight, volume, power, scale range, and accuracy requirements formed the basic considerations in the initial specification of displays. This was accomplished by supplying general specifications (size, scale range, accuracy requirements) to display designers who explored various solutions to displaying the parameter by sketching alternative ways of presenting the required information. Depending upon the ground rules established as system constraints, these displays ranged from variations of standard displays to more advanced designs. With a preliminary sample of alternative sketches, possible designs were reviewed and promising approaches were selected for further elaboration and detail. Through this process, numerous display designs were reviewed and more stringent display specifications were gradually defined. In short, the population of possible designs for particular display parameters was gradually reduced.

Engineering estimates of power, weight and volume requirements for particular displays may be obtained at most of the stages in this specification process. The estimates derived for the APOLLO control-display subsystem can be considered approximations of what is required with respect to these factors. The reason for this is that detailed instrument drawings are required to provide exact figures. In addition, since current specifications are the result of the completion of only the first cycle of this design method, it is advisable to wait for further iterations before detailing the black boxes which might be behind each display parameter.

The control-display Specification Charts (Sub-section 2. 3) present the specifications for the displays which have been derived for the APOLLO system. Figure II-2-4 shows the resulting control-display subsystem mock-up. Further modification and refinement will take place during the development process.

2.2.5 Display Evaluation and Selection

Due to the time limitations of the APOLLO study, it was not possible to carry out in minute detail all the steps of the design method described in Figure II-2-1. Additional screening of the displays must be conducted to verify their effectiveness. It should be noted that further experimentation is recommended although it was not utilized in support of some of the analytic steps of this method. These requirements are spelled out to indicate that further effort within the framework of the methodological model is necessary before final justification of the control-display subsystem can be obtained.

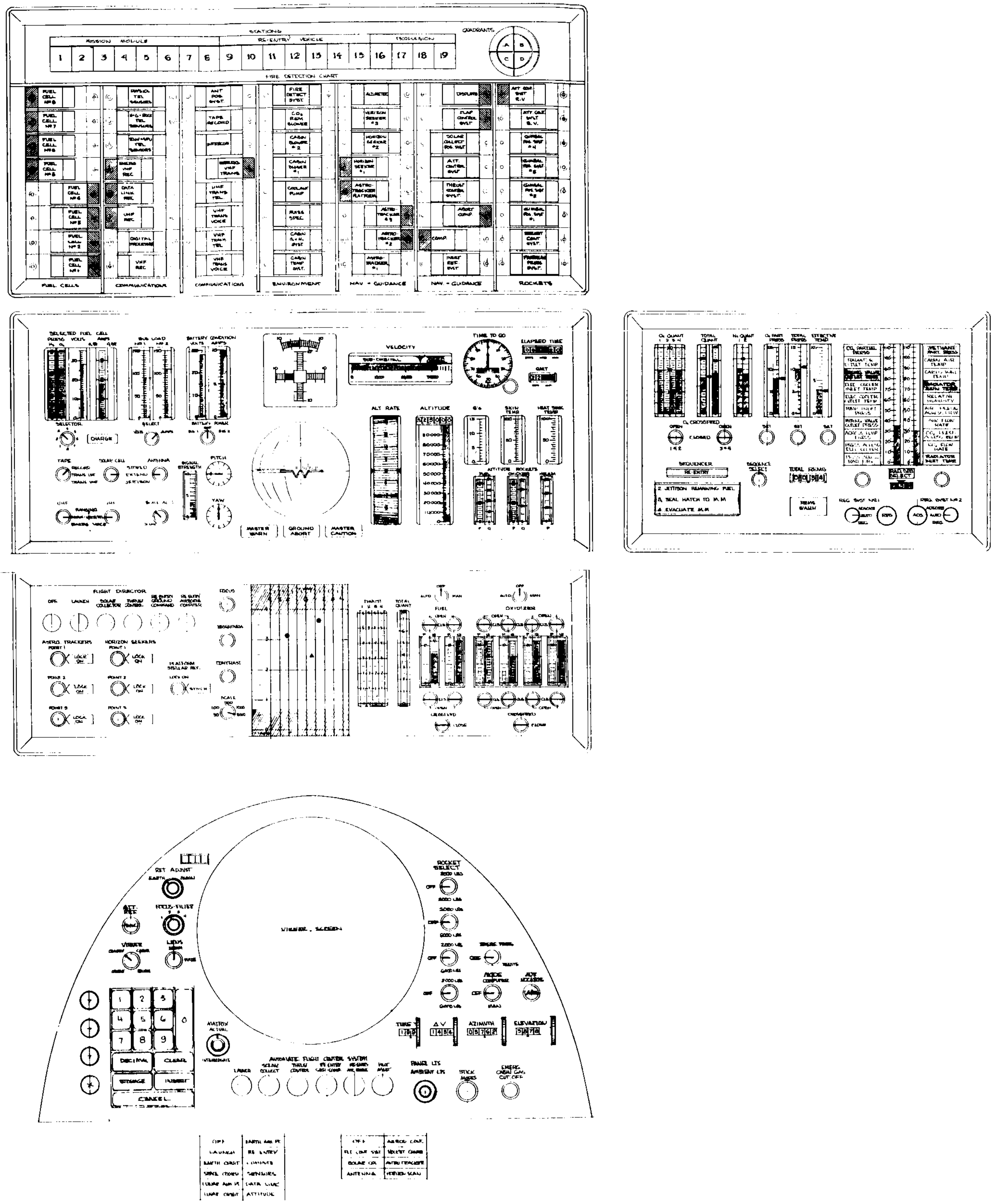


Figure II-2-4. Control - display subsystem

2.3 FUNCTIONAL ALLOCATION AND CONTROL-DISPLAY PARAMETER CHARTS

This sub-section contains Functional Allocation Charts (Tables II-2-I through II-2-V) and Control Display Parameter Charts (Tables II-2-VI through II-2-X).

The Functional Allocation charts present, in summary form for each subsystem, all the vehicle functions which relate to the crew, and describe for each mission phase (1) how the function is allocated to the vehicle system e.g., M = man, G = ground, S = system and (2) the nature of crew participation in this function e.g. (M) monitor or C = control.

For each subsystem the C-D Parameter Charts define the implementation of these functions in terms of hardware. Presented on these charts is the nature of the hardware in terms of types of instrument and control, scales required, accuracy of reading, power, and weight figures, and the relationship of hardware items to one another.

2.4 CREW STATION PANEL LAYOUT

2.4.1 Considerations Regarding Crew Functions and Division of Responsibility

A preliminary definition of the areas of crew member function was presented in the Midterm Report and has been refined as the study proceeded.

From the standpoint of an integrated control-display system, the design objective of providing the crew with equipment to do the job requires definition of: panel area layout, allocation of parameters, and seat arrangement.

Within the requirement that APOLLO will be a three-man vehicle, there arose the question of the composition of the three-man crew and division of function among the crew members. Prime factors which determine the crew composition are workload, training requirements and operational procedures.

When the control tasks were laid out in this study, a re-evaluation was made of the need for three men in the vehicle. There appear to be several good reasons for this number. One concerns the work-rest cycle. With a three man crew, one man can be off duty for an eight hour period, while the remaining crew members work. It is recommended that the rest period be as close to the normal earth eight hours as possible, except for periods where critical vehicle functions must occur. Then the three crew members

[redacted]

TABLE II-2-I — FUNCTIONAL ALLOCATION CHART — NAVIGATION AND GUIDANCE

SUBSYSTEM I — Navigation & Guidance	MISSION PHASE									
FUNCTIONAL ANALYSIS — SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
A. Position & Velocity Parameters										
1. Orbital Parameters (Earth-Moon Reference)										
a. Input Computational Source Selection	(M)	(M)	M	M			M	M		
1. Computer-platform	M	M	M	M			M	M		
2. Data Link	M	M	M	M			M	M		
b. Apogee	SM	M	SM	M			SM	M		
c. Perigee	SM	M	SM	M			SM	M		
d. Inclination Angle	S	M	S	M			S	M		
e. Period	SM	M	SM	M			SM	M		
f. Ascending Node	S	M	S	M			S	M		
g. Central Angle of Vehicle	S	M	S	M			S	M		
h. Central Angle of Perigee	SM	M	SM	M			SM	M		
2. Distance-Velocity Parameters (To Earth - To Moon)										
a. Input Computational Source Selection	M	M	M	M	M	M	M	M	M	M
1. Periscope - Time			M	M	M	M	M	M		
2. Horizon Seekers - Time			M	M	M	M	M	M		
3. Altimeter (Radar) - Time	M	M	M	M	M	M	M	M	M	M
3. Predicted Data - At Some Future Time										
- At Occurence of Some Future Event										
a. Orbital Parameters	S	M	S	M			S	M		
b. Distance-Velocity Parameters (Earth Reference)										
1. Min.-Max. Range as Function of Vehicle capabilities, initial conditions, "G", temperature and skip limits									S	M

[redacted]

TABLE II-2-I — FUNCTIONAL ALLOCATION CHART — NAVIGATION AND GUIDANCE (Cont)

SUBSYSTEM I Navigation & Guidance	MISSION PHASE									
FUNCTIONAL ANALYSIS — SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
2. Landing Location of Vehicle on Earth									S	M
3. Location of Alternate Landing Sites on Earth									S	M
c. Predicted Miss Distance - Velocity Error	S	M			S	M	S	M		
Parameters (From Desired Position and										
Velocity For Initiating Moon Orbit and										
Re-entry) (Earth-Moon Reference)										
1. Input Computational Source Selection	M	M	M	M	M	M	M	M		
a. Computer and Astrotrackers	M	M	M	M	M	M	M	M		
b. Computer and Horizon Seekers	M	M	M	M	M	M	M	M		
c. Theodolite	M	M	M	M	M	M	M	M		
1. Computer Assist	M	M	M	M	M	M	M	M		
2. No Computer Assist	M	M	M	M	M	M	M	M		
d. Data Link	M	M	M	M	M	M	M	M		
2. Predicted Miss Distance and Allowable Limits										
a. X Axis	SM	M			SM	M	SM	M		
b. Y Axis	SM	M			SM	M	SM	M		
c. Z Axis	SM	M			SM	M	SM	M		
d. Re-entry Angle	SM	M	SM	M	SM	M	SM	M		
3. Predicted ΔV Error and Allowable Limits	SM	M			SM	M	SM	M		

[redacted]

TABLE II-2-I — FUNCTIONAL ALLOCATION CHART — NAVIGATION AND GUIDANCE (Cont)

SUBSYSTEM I ▷ Navigation & Guidance	MISSION PHASE									
FUNCTIONAL ANALYSIS — SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
B. Comparison (Checks) of Position & Velocity Parameters										
1. Type of Data to Check, i.e., Orbital Parameters,	M	M	M	M	M	M	M	M		
Distance-Velocity Parameters, Predicted Miss										
Distance-Velocity Parameters										
2. Source of Data For Check, i.e., Astrotrackers	M	M	M	M	M	M	M	M		
Horizon Seekers, Theodolite, Data Link										
Computations, Platform										
3. Corrections (Updating Computer)										
a. Manual or Automatic Input	M	M	M	M	M	M	M	M		
C. Time To:										
1. Initiate "Safest" abort sequency	S	M								
2. Initiate Secondary Mission	S	M								
3. Initiate Moon Orbit							S	M		
4. Initiate Escape From Moon Orbit							S	M		
5. Initiate Mid-course Corrections					M	M				
6. Initiate Re-entry									S	M
7. Release Drogue Chute									SM	M
8. Release Reefed Main Chutes No. 1 and No. 2									SM	M
9. Unreef Main Chutes No. 1 and No. 2									SM	M
10. Release Heat Shield									SM	M
11. Expand Impact Bags									SM	M

[redacted]

TABLE II-2-I — FUNCTIONAL ALLOCATION CHART — NAVIGATION AND GUIDANCE (Cont)

SUBSYSTEM I — Navigation & Guidance	MISSION PHASE									
FUNCTIONAL ANALYSIS — SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
D. Thrust Corrections										
1. Manual or Automatic Insert of Required ΔV					M	M	M	M		
2. Program Selection										
a. Min. Time					M	M	M	M		
b. Min. Fuel					M	M	M	M		
3. Required ΔV										
a. Time to Initiate					M	M	M	M		
b. Magnitude					SM	M	SM	M		
c. Direction					SM	M	SM	M		
d. Thrust Level					M	M	M	M		
E. Attitude Control										
1. 3 Axis Attitude										
a. Present	S	M	SM	M	SM	M	SM	M	SM	M
b. Command	S	M	SM	M	SM	M	SM	M	SM	M
2. 3-Axis Attitude Rates	S	M	SM	M	SM	M	SM	M	SM	M
3. Attitude Reference Source										
a. Platform			M	M	M	M	M	M	M	M
b. Computer			M	M	M	M	M	M	M	M
c. Periscope			M	M	M	M	M	M	M	M
d. Horizon Seekers			M	M	M	M	M	M	M	M

[redacted]

TABLE II-2-I — FUNCTIONAL ALLOCATION CHART — NAVIGATION AND GUIDANCE (Cont)

SUBSYSTEM I ▷ Navigation & Guidance	MISSION PHASE									
FUNCTIONAL ANALYSIS — SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
F. Aerodynamic Control										
1. Present "G" and Limits									SM	M
2. Total Heat Absorbed and Limits									SM	M
3. Present Skin Temperature and Limits									SM	M
4. Skip Limits									SM	M
5. Altitude Rate									SM	M
a. Acceleration									SM	M
6. Obstacle Avoidance and Terrain Clearance									M	M
G. Celestial and Lunar Observations										
1. Periscope			M	M	M	M	M	M		
a. Focus			M	M	M	M	M	M		
b. Lens Select			M	M	M	M	M	M		
c. Filters			M	M	M	M	M	M		
2. TV Cameras			M	M	M	M	M	M		
3. Scientific Observations of Moon's Surface										
a. Spectrometer							M	GM		
b. Camera										
c. 1. Area Mapping							MS	GM		
2. Steroscopic Photo Mapping							MS	GM		

[redacted]

TABLE II-2-II — FUNCTIONAL ALLOCATION CHART — ROCKET POWER

SUBSYSTEM 2 Rocket Power

FUNCTIONAL ANALYSIS

SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM

	MISSION PHASE									
	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
A. Time to Fire	G	MG			SM	M	SM	M		
B. Disconnect Unbilical Cord	G	G								
C. Rocket Engine Condition										
1. Saturn - Stage No. 1 - 8 Engines	S	G								
Stage No. 2 - 4 Engines										
Stage No. 3 - 4 Engines										
2. Mission Engines 4 Main Engines										
2 Low Thrust Engines										
a. Pressure					SM	M	SM	M		
b. Condition										
1. TV Observations					M	M	M	M		
c. Gimballed Position					SM	M	SM	M		
D. Rocket Engine Select										
1. Desired Thrust Level					M	M	M	M		
a. Combination of Main and Low Thrust Engines(1-6					M	M	M	M		
E. Fuel and Oxidizer System (Mission Engines)										
(Fuel Tank Compartments for Mid-course - 2)										
(Oxidizer Tank Compartments for Mid-course - 4)										
1. Total Consumed					SM	M	SM	M		
2. Total Remaining					SM	M	SM	M		
3. Select Tanks To Supply Individual Engines					M	M	M	M		
4. Select Tanks to Empty and to Fill					M	M	M	M		

TABLE II-2-II — FUNCTIONAL ALLOCATION CHART — ROCKET POWER (Cont)

SUBSYSTEM 2 Rocket Power	MISSION PHASE									
FUNCTIONAL ANALYSIS SYMBOLS (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON	MID-COURSE TO EARTH	TRANSFER TO MOON	TRANSFER TO EARTH	RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
F. Pressure System (Mission Engines)										
1. Time to Pressurize					M	M	M	M		
2. Fuel and Oxidizer Pressure					SM	M	SM	M		
G. Staging Sequence										
1. Abort										
a. Engine Shutdown (Boost Only)	SMG									
b. Jettison of Escape Capsule	SMG									
c. Abort Rockets - Thrust										
1. Magnitude	Preset	G								
2. Direction	Preset	G								
3. Thrust Level	Preset	G								
2. Secondary Mission - Lunar Mission	S	MG								
H. Jettison of Abort Rockets									SM	M
I. Jettison of Mission Engines									SM	M
J. Jettison of Mission Module									SM	M

[redacted]

TABLE II-2-III — FUNCTIONAL ALLOCATION CHART — COMMUNICATIONS

SUBSYSTEM 3 Communications	MISSION PHASE									
FUNCTIONAL ANALYSIS — SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
A. Transmitters										
1. Equipment Select (Status - Operate Mode)	M	M	M	M	M	M	M	M	M	M
a. UHF - Tel.	M	M	M	M	M	M	M	M	M	M
b. UHF - Voice	M	M	M	M	M	M	M	M	M	M
c. VHF - Tel.	M	M	M	M	M	M	M	M	M	M
d. VHF - Voice	M	M	M	M	M	M	M	M	M	M
e. Emergency VHF	M	M	M	M	M	M	M	M	M	M
2. Transmit Mode (Signal Strength and Message Backlog)	M	M	M	M	M	M	M	M	M	M
a. Voice (Volume)	M	M	M	M	M	M	M	M	M	M
b. Telemetry (Automatic - Manual)	M	M	M	M	M	M	M	M	M	M
c. Emergency VHF (Key Data Insert)	M	M	M	M	M	M	M	M	M	M
d. Signal Strength Meter	M	M	M	M	M	M	M	M	M	M
B. Receivers										
1. Equipment Select (On-Off)	M	M	M	M	M	M	M	M	M	M
a. VHF	M	M	M	M	M	M	M	M	M	M
b. UHF	M	M	M	M	M	M	M	M	M	M
c. VHF (Emergency)	M	M	M	M	M	M	M	M	M	M
d. Data Link	S	M	S	M	S	M	S	M	S	M
2. Receive Mode										
a. Voice (Volume)	M	M	M	M	M	M	M	M	M	M
b. Tel. Encoder (On-Off)	M	M	M	M	M	M	M	M	M	M

[redacted]

[redacted]

TABLE II-2-III — FUNCTIONAL ALLOCATION CHART — COMMUNICATIONS (Cont)

SUBSYSTEM 3 Communications	MISSION PHASE									
FUNCTIONAL ANALYSIS SYMBOLS (C) = CONTROL G = GROUND (M) = MONITOR M = MAN S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
C. Telemetry Sensors (On-Off)										
1. Env. and S.P.U.	M	M	M	M	M	M	M	M	M	M
2. Nav. and Guid. Rockets	M	M	M	M	M	M	M	M	M	M
3. Physiological Data	M	M	M	M	M	M	M	M	M	M
D. Intercom (On-Off)	M	M	M	M	M	M	M	M	M	M
E. Tape Recorder	M	M	M	M	M	M	M	M	M	M
F. High Gain Antenna										
1. Rotational Control 90° - 2 Axis	M	M	M	M	M	M	M	M	M	M
G. Malfunctions		M		M		M		M		M
H. Erecting Antenna (Re-entry)									M	MG
I. Chaff Ejector									M	MG
J. Flare Cluster									M	MG
K. Sofar Bomb									M	MG
L. Dye Marker									M	MG

[redacted]

TABLE II-2-IV — FUNCTIONAL ALLOCATION CHART — SECONDARY POWER

SUBSYSTEM 4 Secondary Power	MISSION PHASE									
FUNCTIONAL ANALYSIS — SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
A. Input Form Solar Modules (8)	S	M	S	M	S	M	S	M		
B. Capacity - Fuel Cell Batteries (8)										
1. Hydrogen Pressure		M		M		M		M		
2. Oxygen Pressure		M		M		M		M		
C. Output of Fuel Cell Batteries - Solar Modules										
1. Voltage	S	M	S	M	S	M	S	M		
2. Current	S	M	S	M	S	M	S	M		
D. Voltage/Current										
1. Buss No. 1	M	M	M	M	M	M	M	M	M	M
2. Buss No. 2	M	M	M	M	M	M	M	M	M	M
E. Switching (Circuit Breakers)	M	M	M	M	M	M	M	M	M	M
1. Fuel Cell Batteries to Buss No. 1 or No. 2	M	M	M	M	M	M	M	M	M	M
2. Functions - From Buss No. 1 or No. 2	M	M	M	M	M	M	M	M	M	M
a. Communications Subsystem	M	M	M	M	M	M	M	M	M	M
1. Emergency VHF Receiver	M	M	M	M	M	M	M	M	M	M
2. UHF Receiver	M	M	M	M	M	M	M	M	M	M
3. Emergency VHF Transmitter	M	M	M	M	M	M	M	M	M	M
4. UHF - Tel	M	M	M	M	M	M	M	M	M	M
5. UHF Transmitter - Voice	M	M	M	M	M	M	M	M	M	M
6. VHF Transmitter - Tel	M	M	M	M	M	M	M	M	M	M
7. VHF Transmitter - Voice	M	M	M	M	M	M	M	M	M	M
8. Antenna Positioning System	M	M	M	M	M	M	M	M	M	M
9. Tape Recorder	M	M	M	M	M	M	M	M	M	M
(Continued)										

TABLE II-2-IV — FUNCTIONAL ALLOCATION CHART — SECONDARY POWER (Cont)

SUBSYSTEM 4 Secondary Power — FUNCTIONAL ANALYSIS (SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM)	LAUNCH (C)	LAUNCH (M)	SECONDARY MISSION EARTH ORBIT (C)	SECONDARY MISSION EARTH ORBIT (M)	MID-COURSE TO MOON EARTH (C)	MID-COURSE TO MOON EARTH (M)	TRANSFER TO MOON EARTH (C)	TRANSFER TO MOON EARTH (M)	RE-ENTRY (C)	RE-ENTRY (M)
10. Intercom	M	M	M	M	M	M	M	M	M	M
11. Data Link Receiver	M	M	M	M	M	M	M	M	M	M
12. VHF Receiver	M	M	M	M	M	M	M	M	M	M
13. Telemetry Encoder	M	M	M	M	M	M	M	M	M	M
14. Tel. Sensors (Environment and SPU)	M	M	M	M	M	M	M	M	M	M
15. Tel. Sensors (Navigation, Guidance and	M	M	M	M	M	M	M	M	M	M
Rockets)										
16. Tel. Sensors (Physiological)	M	M	M	M	M	M	M	M	M	M
b. Environment Subsystem	M	M	M	M	M	M	M	M	M	M
1. Cabin Temperature System	M	M	M	M	M	M	M	M	M	M
2. Cabin O_2 System	M	M	M	M	M	M	M	M	M	M
3. Cabin N_2 System	M	M	M	M	M	M	M	M	M	M
4. Coolant Pumps	M	M	M	M	M	M	M	M	M	M
5. Cabin Blower No. 1	M	M	M	M	M	M	M	M	M	M
6. Cabin Blower No. 2	M	M	M	M	M	M	M	M	M	M
7. CO_2 Removal Blower	M	M	M	M	M	M	M	M	M	M
8. Cabin Fans	M	M	M	M	M	M	M	M	M	M
9. Fire Detection System	M	M	M	M	M	M	M	M	M	M

TABLE II-2-IV — FUNCTIONAL ALLOCATION CHART — SECONDARY POWER (Cont)

SUBSYSTEM 4 Secondary Power	MISSION PHASE									
FUNCTIONAL ANALYSIS — SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON	MID-COURSE TO EARTH	TRANSFER TO MOON	TRANSFER TO EARTH	RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
c. Navigation and Guidance Subsystem	M	M	M	M	M	M	M	M	M	M
1. Astrotracker No. 1	M	M	M	M	M	M	M	M	M	M
2. Astrotracker No. 2	M	M	M	M	M	M	M	M	M	M
3. Astrotracker No. 3	M	M	M	M	M	M	M	M	M	M
4. Astrotracker Platform	M	M	M	M	M	M	M	M	M	M
5. Horizon Seeker No. 1	M	M	M	M	M	M	M	M	M	M
6. Horizon Seeker No. 2	M	M	M	M	M	M	M	M	M	M
7. Horizon Seeker No. 3	M	M	M	M	M	M	M	M	M	M
8. Altimeter	M	M	M	M	M	M	M	M	M	M
9. Inertial Reference	M	M	M	M	M	M	M	M	M	M
10. Computer	M	M	M	M	M	M	M	M	M	M
11. Abort Computer	M	M	M	M	M	M	M	M	M	M
12. Thrust Control System	M	M	M	M	M	M	M	M	M	M
13. Attitude Control System	M	M	M	M	M	M	M	M	M	M
14. Solar Collector Positioning System	M	M	M	M	M	M	M	M	M	M
15. Flap Control System	M	M	M	M	M	M	M	M	M	M
16. Displays	M	M	M	M	M	M	M	M	M	M

TABLE II-2-IV — FUNCTIONAL ALLOCATION CHART — SECONDARY POWER (Cont)

SUBSYSTEM 4 Secondary Power

FUNCTIONAL ANALYSIS SYMBOLS (C) = CONTROL G = GROUND (M) = MONITOR M = MAN S = SYSTEM	MISSION PHASE: LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
d. Rocket Power Subsystem	M	M	M	M	M	M	M	M	M	M
1. Propellant Pressurization System	M	M	M	M	M	M	M	M	M	M
2. Rocket Control System	M	M	M	M	M	M	M	M	M	M
3. Gimbal Pos. System No. 1	M	M	M	M	M	M	M	M	M	M
4. Gimbal Pos. System No. 2	M	M	M	M	M	M	M	M	M	M
5. Gimbal Pos. System No. 3	M	M	M	M	M	M	M	M	M	M
6. Gimbal Pos. System No. 4	M	M	M	M	M	M	M	M	M	M
7. Attitude Control System (S.V.)	M	M	M	M	M	M	M	M	M	M
8. Attitude Control System (R.V.)	M	M	M	M	M	M	M	M	M	M
e. Secondary Power	M	M	M	M	M	M	M	M	M	M
1. Fuel Cell No. 1	M	M	M	M	M	M	M	M	M	M
2. Fuel Cell No. 2	M	M	M	M	M	M	M	M	M	M
3. Fuel Cell No. 3	M	M	M	M	M	M	M	M	M	M
4. Fuel Cell No. 4	M	M	M	M	M	M	M	M	M	M
5. Fuel Cell No. 5	M	M	M	M	M	M	M	M	M	M
6. Fuel Cell No. 6	M	M	M	M	M	M	M	M	M	M
7. Fuel Cell No. 7	M	M	M	M	M	M	M	M	M	M
8. Fuel Cell No. 8	M	M	M	M	M	M	M	M	M	M

TABLE II-2-V — FUNCTIONAL ALLOCATION CHART — ENVIRONMENT

SUBSYSTEM 5 Environment	MISSION PHASE									
FUNCTIONAL ANALYSIS SYMBOLS (C) = CONTROL G = GROUND (M) = MONITOR M = MAN S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
A. Lighting										
1. Location	M	M	M	M	M	M	M	M	M	M
2. Intensity	M	M	M	M	M	M	M	M	M	M
B. Air Conditioning (Maintain "Effective" Temperature)	M	M	M	M	M	M	M	M	M	M
1. Gas Supply (Primary) (Flow Rate and Quantity)	MS	M	MS	M	MS	M	MS	M	MS	M
a. LOX										
1. Tank No. 1	M	M	M	M	M	M	M	M	M	M
2. Tank No. 2	M	M	M	M	M	M	M	M	M	M
3. Tank No. 3	M	M	M	M	M	M	M	M	M	M
b. Diluent N_2										
1. Tank No. 1	M	M	M	M	M	M	M	M	M	M
2. Gas Supply (Emergency) (Flow Rate and Quantity)	MS	M	MS	M	MS	M	MS	M	MS	M
a. LOX - Tank No. 4	M	M	M	M	M	M	M	M	M	M
b. Diluent N_2 - Tank No. 2	M	M	M	M	M	M	M	M	M	M
3. Supply Switching										
a. Select Tanks to Empty and To Fill	M	M	M	M	M	M	M	M	M	M

[redacted]

TABLE II-2-V — FUNCTIONAL ALLOCATION CHART — ENVIRONMENT (Cont)

SUBSYSTEM 5 Environment		MISSION PHASE									
FUNCTIONAL ANALYSIS	SYMBOLS (C) = CONTROL G = GROUND (M) = MONITOR M = MAN S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
		(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
B. Air Conditioning (Continued from Page 1)											
4. Temperature (Atmospheric and Wall Temperature)			M		M		M		M		M
a. Blower No. 1 and/or Blower No. 2		M	M	M	M	M	M	M	M	M	M
b. Coolant Fluid System											
1. Pump		S	M	S	M	S	M	S	M	S	M
2. Radiator - Cools Fluid by Radiation		S	M	S	M	S	M	S	M	S	M
3. Mixing Valve (Thermostat)		M	M	M	M	M	M	M	M	M	M
4. Cabin Heat Exchanger		S	M	S	M	S	M	S	M	S	M
5. Electronics Heat Exchanger		S	M	S	M	S	M	S	M	S	M
c. Relative Humidity											
1. H_20 Collector		S	M	S	M	S	M	S	M	S	M
2. H_20 Tank		S	M	S	M	S	M	S	M	S	M
5. Circulation Fans		M	M	M	M	M	M	M	M	M	M
C. Cabin Pressure											
1. Total Pressure		SM	M	SM	M	SM	M	SM	M	SM	M
2. p 0_2		SM	M	SM	M	SM	M	SM	M	SM	M
3. p H_20			M		M		M		M		M
4. p $C0_2$		SM	M	SM	M	SM	M	SM	M	SM	M
5. Leak detection (Oxygen Consumption Rate)		SM	M	SM	M	SM	M	SM	M	SM	M

[redacted]

[redacted]

TABLE II-2-V — FUNCTIONAL ALLOCATION CHART — ENVIRONMENT (Cont)

SUBSYSTEM 5 Environment	MISSION PHASE									
FUNCTIONAL ANALYSIS SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
D. Noxious Gas (Toxic Included)										
1. Sensors (Mass Spectrometer)		MG		MG		MG		MG		MG
2. Removal System - Noxious Gases										
a. Blower No. 3	S		S		S		S		S	
b. Moisture Absorption and Transfer Unit	S		S		S		S		S	
c. Molecular Sieve Bed No. 1 or No. 2	S		S		S		S		S	
1. Removal of noxious gases	S		S		S		S		S	
2. Reactivation of beds	SM	M	SM	M	SM	M	SM	M	SM	M
d. Heat Absorption and Transfer Unit	S		S		S		S		S	
E. Emergency System (Cocoon No. 1, No. 2, No. 3)	M	M	M	M	M	M	M	M	M	M
1. 100% Oxygen	S	M	S	M	S	M	S	M	S	M
2. Effective Temperature System	M	M	M	M	M	M	M	M	M	M
F. Particulate Matter and Bacterial Control										
1. Particulate Matter Filters	S		S		S		S		S	
G. Fire Detection and Control System										
1. Warning and Localization Signals	S	M	S	M	S	M	S	M	S	M
H. Radiation Detection System (Rate)	S	M	S	M	S	M	S	M	S	M

[redacted]

[redacted]

TABLE II-2-VI — CONTROL-DISPLAY PARAMETERS CHART — NAVIGATION AND GUIDANCE

PHASES.....
1. LAUNCH
2. TRANSFER ORBIT
3. LUNAR ORBIT
4. RE-ENTRY

CONTROL - DISPLAY PARAMETERS

SUBSYSTEM 1
PAGE 1 OF 2

NAVIGATION AND GUIDANCE

ITEM NO.	PARAMETERS	INSIDE COCOON	REQUIRED DURING PHASES 1	2	3	4	CONTROL MODES	FUNCTIONALLY RELATED TO. SUB. SYS.	ITEM	RANGE	DISPLAY: MOVING ELEMENT TAPE	POINTER	COUNTER	FACE HEIGHT INCHES	WIDTH INCHES	DEPTH INCHES	WEIGHT POUNDS	POWER WATTS DC	AC	
1.	BODY AXIS RATES - YAW, PITCH, ROLL		X	X	X	X		1	2-4, 38	0 ± 10°/SEC				3	3	3	.9		9	
2.	VEHICLE ATTITUDE AZIMUTH		X	X	X	X		1	1, 38	0-360°			INCLUDED IN ITEM 4							MOVING SPHERE DISPLAY WITH
3.	ELEVATION		X	X	X	X		1	1, 38	0 ± 90°										COMMAND NEEDLES
4.	ROLL		X	X	X	X		1	1, 38	0 ± 180°				5	5	10	8	-	10	
5.	ALTITUDE RATE					X		1	6-11, 38	0 ± 3500/FT/SEC			SINGLE DISPLAY							LIMITS ARE A FUNCTION OF FLIGHT
6.	ALTITUDE RATE LIMITS					X		1	5-11, 38											PATH ANGLE AND DYNAMIC PRESSURE
7.	ALTITUDE		X			X		1	5-11, 38	0 - 400,000 FT										
8.	LONGITUDINAL "G"					X		1	5-11, 38	0 - 15 G										
9.	SKIN TEMPERATURE					X		1	5-11, 38	0% - 100%			INCLUDED IN ITEM 14							
10.	HEAT SINK TEMPERATURE					X		1	5-11, 38	0% - 100%										
11.	HORIZONTAL VELOCITY					X		1	5-10, 38	0 - 36,000 FPS										
12.	GREENWICH MEAN TIME		X	X	X	X		1	15	0 - 24 HR										
13.	ELAPSED TIME		X	X	X	X		1	15	0 - 14 DAYS										
14.	TIME TO GO			X	X	X		1	15, 47	0 - 100 MIN				9	9	4	8	-	20	
15.	COMPUTER READOUT: LAUNCH MONITOR		X					1	26, 27					4	6	10	20	-	40	
16.	EARTH ORBIT		X			X														READOUT IS ON A CATHODE RAY
17.	SPACE COORDINATES			X	X															TUBE. MODE SELECT IS ON COM-
18.	LUNAR AIM POINT			X																MANDER'S CONSOLE INSIDE COCOON.
19.	LUNAR ORBIT				X															ACTUAL, OR INTERROGATE VALUES
20.	EARTH AIM POINT			X																MAY BE OBTAINED WITH ANY OF THE
21.	RE-ENTRY		X			X														MODES LISTED. DATA MAY ALSO BE
22.	SENSORS			X	X															INSERTED BY MEANS OF A KEYBOARD
23.	DATA LINK			X	X	X														INCLUDED IN ITEM 26.
24.	ATTITUDE			X	X	X														
25.	COMPUTE			X	X	X														
26.	COMPUTER READOUT MODE SELECTOR	X	X	X	X	X	ITEMS 15-25							3	6	2	2	-	-	
27.	COMPUTER READOUT BRIGHT-FOCUS-CONTRACT		X	X	X	X	0 - 10	1	15											
28.	ASTROTRACKER MODE (3 REQUIRED)			X	X		POINT LOCK ON	1	22				INCLUDED IN ITEM 32							
29.	HORIZON SEEKER MODE (3 REQUIRED)			X	X	X	POINT LOCK ON	1	22											
30	RE-ENTRY DISPLAY SCALE CHANGE					X		1	21											2000, 1000, 500, 100, 50 MILES/INCH

[redacted]

TABLE II-2-VI — CONTROL-DISPLAY PARAMETERS CHART — NAVIGATION AND GUIDANCE (Continued)

PHASES.....
1. LAUNCH
2. TRANSFER ORBIT
3. LUNAR ORBIT
4. RE-ENTRY

CONTROL - DISPLAY PARAMETERS

SUBSYSTEM 1
PAGE 2 OF 2
NAVIGATION AND GUIDANCE

ITEM NO.	PARAMETERS	INSIDE COCOON	REQUIRED DURING PHASES 1	2	3	4	CONTROL MODES	FUNCTIONALLY RELATED TO. SUB. SYS.	ITEM	DISPLAY RANGE	MOVING ELEMENT TAPE	POINTER	COUNTER	FACE HEIGHT INCHES	WIDTH INCHES	DEPTH INCHES	WEIGHT POUNDS	POWER WATTS DC	AC	
31	PLATFORM STELLAR REFERENCE MODE SELECT			X	X		SYNCH LOCK ON	1	24		INCLUDED IN ITEM 32									
32	COMMAND NEEDLE MODE SELECT		X	X	X	X	SEE ITEM 48	1	2,3,4					6	9	3	4	-	-	
33	PERISCOPE:	X	X	X	X	X								10	10	-	50	-	-	
34	ALTITUDE RETICLE SET	X	X	X	X	X	MOON EARTH	1	33	0 - 1000 MI										
35	LENS SELECTOR	X	X	X	X	X	WIDE ANGLE NORMAL	1	33											
36	FILTER SELECTOR	X	X	X	X	X	1,2,3,4	1	33											
37	MODE SELECTOR	X	X	X	X	X		1	33											VIEWING-ORIENTATION-COMPUTER-ROCKETS
38	SIDE STICK FLIGHT CONTROLS	X		X	X	X		1	1-4											3 AXIS CONTROLLER
39	AERODYNAMIC CONTROLS					X		1	1-11											
40	SOLAR COLLECTOR			X	X															
41	ANTENNA			X	X															
42	ROCKET CHAMBERS			X	X															
43	ASTROTRACKERS			X	X															
44	HORIZON SCANNERS			X	X	X														
45	LEFT ARMREST CONTROLS ABORT	X	X				OFF ON													
46	SECONDARY MISSION	X	X				OFF ON													
47	SAFETY SWITCH	X		X	X	X	OFF ON													
48	FLIGHT CONTROL MODE SELECT	X	X	X	X	X					INCLUDED IN ITEM 33									ITEMS 32 AND 48 INCLUDE "LAUNCH",
49																				"SOLAR COLLECTOR", "THRUST CONTROL",
50																				"RE-ENTRY AIRBORNE COMPUTER",
																				"RE-ENTRY GROUND COMMAND" AND ITEM
																				48 INCLUDES "PILOT ASSIST", IN
																				ADDITION.

[redacted]

TABLE II-2-VII — CONTROL-DISPLAY PARAMETERS CHART — ROCKET POWER

PHASES.....
1. LAUNCH
2. TRANSFER ORBIT
3. LUNAR ORBIT
4. RE-ENTRY

CONTROL - DISPLAY PARAMETERS

SUBSYSTEM 2 — ROCKET POWER

ITEM NO.	PARAMETERS	INSIDE COCOON	REQUIRED DURING PHASES 1	2	3	4	CONTROL MODES	FUNCTIONALLY RELATED TO. SUB. SYS.	ITEM	DISPLAY RANGE	MOVING ELEMENT TAPE	POINTER	COUNTER	FACE HEIGHT INCHES	WIDTH INCHES	DEPTH INCHES	WEIGHT POUNDS	POWER WATTS DC	AC	
1.	THRUST (FOUR REQUIRED)			X	X			2	15	0 - 6000 LB										
2.	FUEL AND OXIDIZER TOTAL QUANTITY			X	X			2	3,5	0 - 7500 FT/SEC										ASSUMES NORMAL SPECIFIC IMPULSE.
3.	OXIDIZER QUANTITY (FOUR REQUIRED)		X	X	X		ON - OFF	2	4	0 - 1600 LB										
4.	OXIDIZER PRESSURE (FOUR REQUIRED)		X	X	X		AUTO,OFF-ON	2	3	0 - 110 PSI										
5.	FUEL QUANTITY (TWO REQUIRED)		X	X	X		ON - OFF	2	6	0 - 400 LB	ELECTROLUMINESCENT									
6.	FUEL PRESSURE (TWO REQUIRED)		X	X	X		AUTO,OFF-ON	2	5	0 - 110 PSI										
7.	FUEL QUANTITY (ATTITUDE CONTROL SYSTEM)		X	X	X			2	8	0 - 40 LB	ITEMS 1 - 11 ARE TOTALED									
8.	FUEL PRESSURE (ATTITUDE CONTROL SYSTEM)		X	X	X			2	7	0 - 110 PSI	UNDER ITEM 11									
9.	OXIDIZER QUANTITY (ATTITUDE CONTROL SYSTEM)		X	X	X			2	10	0 - 260 LB										
10.	OXIDIZER PRESSURE (ATTITUDE CONTROL SYSTEM)		X	X	X		AUTO, OFF-ON	2	9	0 - 110 PSI										
11.	HELIUM PRESSURE (ATTITUDE CONTROL SYSTEM)		X	X	X			2	8, 10	0 - 3000 PSI				6	10	3	6		40	
12.	THRUST CHAMBER POSITION YAW			X	X					± 5°										MAY BE CONTROLLED MANUALLY THRU COM-
13.	THRUST CHAMBER POSITION PITCH			X	X					± 5°										MANDER'S SIDE STICK & MODE SELECTOR
14.	LOW THRUST ROCKETS (TWO REQUIRED)	X		X	X		OFF ARM													
15.	MAIN ROCKET THRUST SELECT (FOUR REQUIRED)	X	X	X	X		OFF, 2000 LB 6000 LB	2	1											
16.	MAIN ROCKET EMERGENCY PRESSURIZE	X		X	X		OFF - ON	2	2-6											
17.	MAIN ROCKET IGNITION	X	X	X	X		OFF - ON	2	1											
18.	THRUST CONTROL MODE SELECT	X		X	X		COMP. OFF MANUAL	2	15		ITEMS 14 - 22 ARE TOTALED									
19.	THRUST CONTROL MANUAL INPUTS: AZIMUTH	X		X	X		0 - 360°	3			UNDER ITEM 22									THESE ALLOW GROUND SUPPLIED
20.	ELEVATION	X		X	X		0 ± 90°	3												COMMANDS TO BE EXECUTED
21.	MAGNITUDE	X		X	X		0 - 3000 FT/SEC	3												WITHOUT THE AID OF THE ON BOARD
22.	TIME	X		X	X		0 - 60 MIN	3									5	-	-	COMPUTER.
23.	OVERRIDE ROCKET CUTOFF	X		X	X		ON - OFF	2	1											ON LEFT HAND GRIP.
24.																				
25.																				
26.																				
27.																				
28.																				
29.																				

[redacted]

TABLE II-2-VIII — CONTROL-DISPLAY PARAMETERS CHART — COMMUNICATIONS

PHASES.....
1. LAUNCH
2. TRANSFER ORBIT
3. LUNAR ORBIT
4. RE-ENTRY

CONTROL - DISPLAY PARAMETERS

SUBSYSTEM 3 — COMMUNICATIONS

ITEM NO.	PARAMETERS		INSIDE COCOON	REQUIRED DURING PHASES 1	2	3	4	CONTROL MODES	FUNCTIONALLY RELATED TO: SUB. SYS.	ITEM	RANGE	DISPLAY: MOVING ELEMENT: TAPE	POINTER	COUNTER	FACE: HEIGHT INCHES	WIDTH INCHES	DEPTH INCHES	WEIGHT POUNDS	POWER WATTS: DC	AC	COMMUNICATIONS
1.	TRANSMITTERS - VHF VOICE			X			X	ON OFF	3	17											
2.	VHF TELEMETRY			X			X	ON OFF	3	16,17											
3.	UHF VOICE				X	X		ON OFF	3	18											
4.	UHF TELEMETRY				X	X		ON OFF	3	16,18											
5.	VHF EMERGENCY			X	X	X	X	ON OFF	3	24											ITEMS 1 THRU 15 ARE MULTIFUNCTION
6.	RECEIVERS - VHF			X			X	ON OFF													SWITCHES; THE WEIGHT AND SIZE
7.	UHF				X	X		ON OFF													FIGURES FOR WHICH ARE INCLUDED IN
8.	VHF EMERGENCY			X	X	X	X	ON OFF													ITEM 12 UNDER SECONDARY POWER,
9.	DATA LINK			X	X	X	X	ON OFF													"POWER DISTRIBUTION PANEL".
10.	TAPE RECORDER					X		ON OFF	3	16											
11.	INTERCOM			X	X	X	X	ON OFF													
12.	DIGITAL PROCESSER			X	X	X	X	ON OFF	3	13-15											
13.	TELEMETRY SENSORS ENV. & S.P.U.			X	X	X	X	ON OFF	3	12											
14.	N & G, ROCKETS			X	X	X	X	ON OFF	3	12											
15.	PHYSIOLOGICAL			X	X	X	X	ON OFF	3	12											
16.	TAPE RECORDER MODE					X			3	2,4,10											RECORD TRANS. UHF, TRANS. VHF.
17.	TRANSMITTER MODE VHF			X			X		3	1,2											NORMAL-VOICE/TELEM., RANGING,
18.	UHF				X	X			3	3,4											EMERGENCY-VOICE
19.	SIGNAL STRENGTH				X	X	X		3	20,21	0 - 10				INCLUDED IN ITEM 23						
20.	SIGNAL STRENGTH METER ADJUST				X	X	X		3	18											
21.	ANTENNA POSITION ELEVATION				X	X	X	UP-OFF-DN	3	18	0 ± 90°		X								CONTROL IS BY MEANS OF MODE SELEC-
22.	AZIMUTH				X	X	X	LFT-OFF-RT	3	18	0 - 360°		X								TOR AND COMMANDER'S SIDE STICK.
23.	ANTENNA CONTROL				X		X		5	11					3.5	9	3	3	-	5	OFF, EXTEND, JETTISON
24.	CODE KEY				X	X	X	ON-OFF	3	5											
25.																					
26.																					
27.																					
28.																					
29.																					

[redacted]

TABLE II-2-IX — CONTROL-DISPLAY PARAMETERS CHART — SECONDARY POWER

CONTROL - DISPLAY PARAMETERS

SUBSYSTEM 4 — SECONDARY POWER

PHASES.....
1. LAUNCH
2. TRANSFER ORBIT
3. LUNAR ORBIT
4. RE-ENTRY

ITEM NO.	PARAMETERS		INSIDE COCOON	REQUIRED DURING PHASES 1	2	3	4	CONTROL MODES	FUNCTIONALLY RELATED TO: SUB. SYS.	ITEM	DISPLAY: RANGE	MOVING ELEMENT: TAPE	POINTER	COUNTER	FACE: HEIGHT INCHES	WIDTH INCHES	DEPTH INCHES	WEIGHT POUNDS	POWER WATTS: DC	AC	
1.	HYDROGEN PRESSURE (FUEL CELL BATTERY)			X	X	X	X		4	3	0 - 150 PSI		ITEMS 1-9 ARE								ITEMS 1 - 6 MAY BE READ FOR ANY ONE OF THE 8 FUEL CELL BATTERIES
2.	OXYGEN PRESSURE (FUEL CELL BATTERY)			X	X	X	X		4	3	0 - 150 PSI		ELECTROLUMINESCENT.								BY MEANS OF A SELECTOR.
3.	CURRENT TO/FROM FUEL CELL BATTERY			X	X	X	X	CHARGE DISCHARGE	4	1-5	0 - 35 AMP		SIZE - WEIGHT - POWER								THESE ARE ON A SINGLE SCALE WITH
4.	HYDROGEN PRESSURE RATE			X	X	X	X		4	3,1	--		ARE TOTALED UNDER								THE PRESSURE RATE CALIBRATED TO EQUAL THE CURRENT READING IF NO
5.	OXYGEN PRESSURE RATE			X	X	X	X		4	3,2	--		ITEM 9.								LEAKS EXIST.
6.	VOLTAGE (FUEL CELL BATTERY)			X	X	X	X		4	1,2	0 - 35 VOLTS										
7.	BUSS #1 CURRENT/VOLTAGE			X	X	X	X		4		0 - 35 VOLTS										THESE DISPLAYS NORMALLY READ CURRENT. VOLTAGE DISPLAY
8.	BUSS #2 CURRENT/VOLTAGE			X	X	X	X		4		0 - 150 AMP										IS SELECTED WITH A SPRING LOADED SWITCH.
9.	EMERGENCY BATTERY CURRENT/VOLTAGE			X	X	X	X		4	12,15	0 - 30 VOLTS 0 - 10 AMP				5	9	3	5	-	25	
10.	SOLAR CELL OUTPUT				X	X					0 - 10										SHARED WITH ITEM 19, COMMUNICATIONS.
11.	SOLAR COLLECTOR CONTROL			X			X	EXTEND JETTISON	5	11			INCLUDED IN ITEM 9								
12.	POWER DISTRIBUTION PANEL			X	X	X	X	BUSS 1, OFF, BUSS 2	5	1-8					12	24	4	20	-	-	THIS INCLUDES SWITCHES IN ITEMS 13 & 14 PLUS ALL SUBSYSTEM POWER
13.	FUEL CELL BATTERY OUTPUT (8 REQUIRED)			X	X	X	X		5	1-8											SWITCHES SHOWN ON OTHER CHARTS.
14.	SOLAR COLLECTOR POSITIONING SYSTEM				X	X			5	10											
15.	EMERGENCY BATTERY			X	X	X	X	BUSS 1, OFF, BUSS 2	5	7-9			INCLUDED IN ITEM 9								
16.																					
17.																					
18.																					
19.																					
20.																					
21.																					
22.																					
23.																					
24.																					
25.																					
26.																					
27.																					
28.																					
29.																					

[redacted]

TABLE II-2-X — CONTROL-DISPLAY PARAMETERS CHART — ENVIRONMENT

CONTROL - DISPLAY PARAMETERS

PHASES.....
1. LAUNCH
2. TRANSFER ORBIT
3. LUNAR ORBIT
4. RE-ENTRY

SUBSYSTEM 5 — ENVIRONMENT

ITEM NO.	PARAMETERS	INSIDE COCOON	REQUIRED DURING PHASES 1	2	3	4	CONTROL MODES	FUNCTIONALLY RELATED TO. SUB. SYS.	ITEM	RANGE	DISPLAY — MOVING ELEMENT TAPE	POINTER	COUNTER	FACE HEIGHT INCHES	WIDTH INCHES	DEPTH INCHES	WEIGHT POUNDS	POWER WATTS DC	AC	
1.	TOTAL PRESSURE		X	X	X	X	SET 0-15			0-15 PSI										
2.	OXYGEN PARTIAL PRESSURE		X	X	X	X	SET 3-10			0-10 PSI										
3.	EFFECTIVE TEMPERATURE		X	X	X	X	SET 0-10	5		0-10°										
4.	OXYGEN QUANTITY TANKS 1,2,3,4		X	X	X	X		5	1,2,6	0-30 LB		ELECTROLUMINESCENT								SIZE, WEIGHT AND POWER
5.	NITROGEN QUANTITY TANKS 1, 2		X	X	X	X		5	1,6	0-12 LB										INCLUDED IN ITEM 8
6.	TOTAL QUANTITY VS TIME O_2 AND N_2		X	X	X	X		5	4,5	0-15 DAYS										
7.	CABIN GAS CUTOFF	X		X	X	X	ON - OFF	5	1											
8.	OXYGEN CROSSFEED FROM ROCKET TANKS			X	X		ON - OFF	5	4,6					5	8	3	6	-	30	
9.	RADIATION RATE AND TOTALIZER			X	X					0-30 REMS			X	2	3	3	1.2	-	3	
10.	CO_2 REMOVAL SYSTEM NO. 1 AND 2 MODE			X	X	X		5	23,24	ADSORB REGENERATE		INCLUDED IN ITEM 12								ADSORB, AUTO, REGENERATE
11.	SEQUENCING DISPLAY AND MODE SELECT		X	X	X	X					X			3	4	3	.5	-	-	LAUNCH, LUNAR ORBIT ENTRY-EXIT, RE-ENTRY
12.	ENVIRONMENT SYSTEM INTERROGATION DISPLAY		X	X	X	X						X		8.5	5.5	3	4	-	5	
13.	RADIATOR INLET TEMPERATURE			X	X			5	3	50° - 200°F										
14.	RADIATOR OUTLET TEMPERATURE			X	X			5	3	0° - 100°F										
15.	MIXING VALVE OUTLET TEMPERATURE			X	X			5	3	0° - 100°F										
16.	ELECTRONIC COOLER INLET TEMPERATURE			X	X					0° - 100°F										
17.	ELECTRONIC COOLER OUTLET TEMPERATURE			X	X					0° - 200°F										
18.	PUMP INLET PRESSURE			X	X					0-10 PSI										
19.	MIXING VALVE OUTLET PRESSURE			X	X					0-10 PSI										
20.	PRESSURE ACROSS PUMP			X	X					0-10 PSI										SIZE, WEIGHT AND POWER ARE
21.	PRESSURE ACROSS ELECTRONIC COOLER			X	X					0 - 5 PSI										INCLUDED IN ITEM 12.
22.	PRESSURE ACROSS RADIATOR SEGMENTS 1-8			X	X					0 - 1 PSI										THESE ITEMS ARE TIME
23.	CO_2 PARTIAL PRESSURE		X	X	X	X		5	10	0-.3 PSI										SHARED ON TWO READOUTS.
24.	CO_2 ABSORBER FLOW RATE		X	X	X	X		5	10	0-100 CFM										
25.	CABIN AIR FLOW RATE		X	X	X	X				0-150 CFM										
26.	PRESSURE ACROSS CO_2 REMOVAL BLOWER		X	X	X	X		5	10	0-5" H_2O										
27.	PRESSURE ACROSS CABIN AIR BLOWER		X	X	X	X				0-5" H_2O										
28.	RADIATOR SKIN TEMPERATURE			X	X			5	3	0° - 100°F										
29.	CABIN WALL TEMPERATURE		X	X	X	X		5	3	0° - 200°F										

must all be at their vehicle stations. It is apparently quite feasible to shift the "phase" of the period of maximum daily alertness, but it is difficult to alter the length of the normal 24 hour sleep-wakefulness cycle. Thus, it would be possible to distribute the periods of maximum wakefulness of the crew members while maintaining the duration of the sleep periods, so as to cover each 24 hour period as evenly as possible.

The additional reasons for a three man crew relate to the task loading and factors of reliability. It appears that one operator could adequately perform normal midcourse monitoring functions. However, during midcourse navigational changes, and during launch and re-entry, the load on a single operator would be too great. To have total command of the vehicle, two men appear to be required. With this tentative task loading, the third crew member supplies the backup capability for successfully completing the mission in the event of one man becoming incapacitated.

There will be times when all three crew members will be on duty at the same time or must be stationed at fixed positions. At the minimum, this will be during launch, during re-entry and landing, and during periods in which course changes are being made. In addition, there may be particular tasks which will require three crew members for execution such as determining midcourse correction and executing the data-gathering when in the vicinity of the Moon. During most of the flight, however, it is expected that only one man will be required to operate the vehicle's control station and the other two will be free for sleeping, housekeeping, scientific and other functions.

The primary control stations, then, must be designed for occupancy by three crew members simultaneously, and provide for division of function among them during the critical control phases. In addition one-man supervision of necessary display and control equipment during the long free-flight periods is required.

The requirement for interchangeability of function among the crew members is very real and suggests immediately that the three crew members be chosen with the same sort of background and initial capabilities. At the same time, however, the critical nature of some functions dictates that there be rigid selection requirements relative to these functions, and sufficient cross training to allow full capability in each of the specialities involved. The question of differential abilities among the crew depends upon an identification of the critical factors which will be required of the crew in accomplishing their mission.

The crew needs to have on board, for reliability purposes, the kind of knowledge which would allow them to take advantage of all possibilities for utilization, repair, and backup of the equipment. To insure maximum capability and flexibility in the area of flight operations, someone on board should have a working knowledge of astrodynamics to be able to utilize astronomical sightings, ground-based computers, air-based computers, slide-rule calculations and quick paper and pencil operations whenever these are needed. This implies a professional level knowledge of astrodynamics and aerodynamic operations. It suggests that the criteria for one individual before beginning training are professional aircraft pilot's experience plus professional level knowledge of astrodynamics. The piloting background will be extremely valuable in providing the astronaut with the experiences of vehicle control, vibration, buffeting, variable g maneuvers etc., and certainly familiarity with flight control instruments and operating procedures. A great deal of additional special training, specifically on the APOLLO system, will also be required.

The vehicle as it leaves the ground will consist of a carefully designed and constructed system utilizing electrical and mechanical equipment. A great deal of this equipment must be in operation throughout the fourteen days of the mission in order to guarantee mission success. At least one individual on board should have an understanding of the equipment function equal to that of the engineers who designed and built it. This very high knowledge level will insure that if there is any possible way the equipment can be kept in operation correctly during the course of the mission, the knowledge of how this is to be done will be available in the vehicle at the appropriate time. Thus, one of the three crew members should have the functional equivalent of a professional level of knowledge in electrical and mechanical engineering.

The specifications just considered do not imply that a third crew member will have no function. Rather they imply that the critical job requirements appear at this point to be defined as stated above. In addition, the requirement for rotation of duty and for the provision for backup function if one of the crew members should be incapacitated, strongly suggests that each of the crew be well versed in these functions. The primary task for each of the crew members sets his area of first responsibility. Extensive cross training will be assumed in order that, insofar as possible, the survival of any one man will be adequate for the continuation and completion of the mission.

Some of the functions of the operation of the vehicle may require simultaneous participation on the part of two or three of the operators. Each crew station will have to be designed in order to allow this. During the long periods of transfer to the Moon, only one of the operators will be on duty at a time. In this case, he will need to be fully competent to interpret the situation being presented by all the displays and to take what control action is necessary.

The division of duties which have been established is as follows: One man will be primarily responsible for the flight operations; the second man will be primarily be responsible for machine conditions; and the third will primarily be responsible for the mission's scientific functions. This breakdown indicates the three-man crew complement would consist of an "astrodynamicist" (the vehicle commander), a "space vehicle engineer" and a "space scientist."

With these functions in mind, tentative seating arrangements were made. These were altered and later refined and confirmed. The resultant arrangement placed the astrodynamicist, i.e. the commander, in a central position. The space scientist was placed on this man's left and the space engineer on his right. Panel areas were then designed which assigned to the astrodynamicist (the principal pilot) operational capability over most of the primary panel area. The space engineer was provided with a limited amount of separate panel area. His functions, many of which are largely monitoring tasks, are designed to be performed by the one man on duty during mid-course. Thus, some of the displays he would use are located to the right of the vehicle commander's panel. Here they can be viewed from either station. Primary responsibility for their use is shifted to one or the other station depending on mission phase and mode of vehicle operation.

No specific display panels are provided within the re-entry vehicle for the third crew member. This is in keeping with the desire to utilize the third man to provide redundancy

[redacted]

in his ability to operate either of the other two crew stations. The scientific instrumentation for which he is primarily responsible is located in the mission module.

2.4.2 Individual Panel Layout

This section of the report is designed to show and describe the specific controls and displays which were provided the crew in order for them to perform their respective functions. This is accomplished by describing, for each subsystem, the panel and the displays located on it.

2.4.2.1 ROCKET POWER PANEL

The lower right corner of the commander's panel contains the thrust and propellant quantity displays, Figure II-2-5. Individual pressurization and cutoff valves are provided for each compartment of the propellant tanks. Separate oxidizer and fuel pressurizing system mode selectors provide individual control and/or manual override of either system. The tanks would normally be left unpressurized to minimize boil-off.

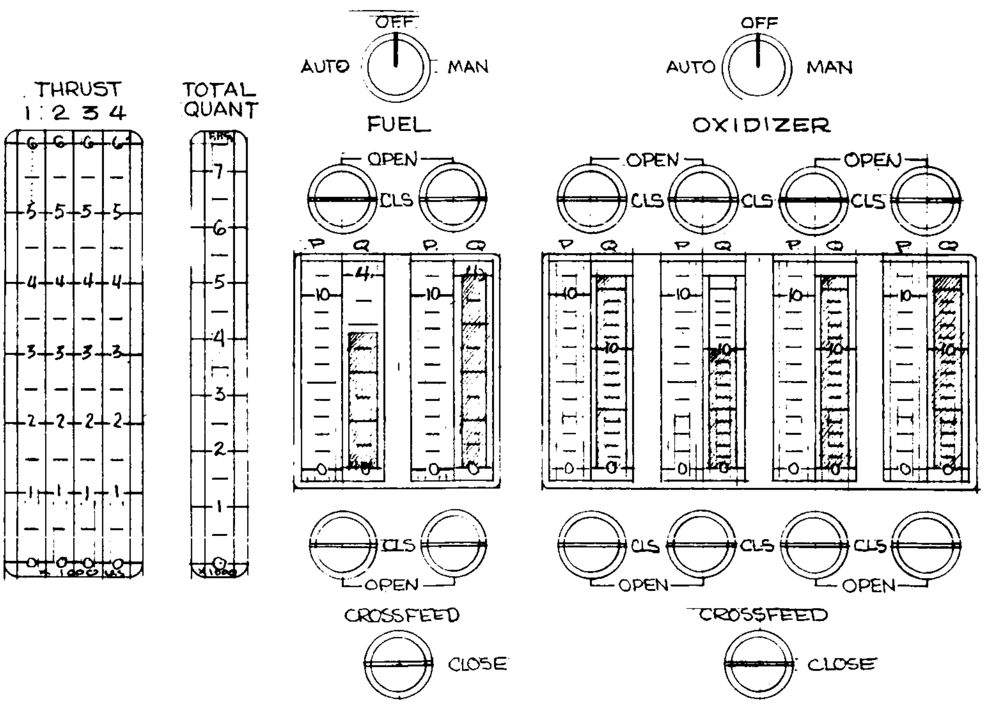


Figure II-2-5. Rocket power panel

[redacted]

Prior to rocket firing the automatic pressurizing mode would be selected and the tank would come up to, and remain at operating pressure. If the pressure fails to rise or overshoots the normal value, the tank pressure may be controlled by actuating the pressurization switch from "off" to "manual". The cutoff valves beneath the quantity and pressure displays allow the crew to use propellant so as to maintain nearly equal quantities in all tanks, thus minimizing the danger of a leaking tank. Of course, rocket engines 1 and 2 will normally operate only from oxidizer tanks 1 and 2 and fuel tank 1. To allow complete flexibility of the system, two crossfeed valves are provided which will allow any engine to be run from any combination of tanks. As an aid to fuel energy management, a total quantity display is provided for both fuel and oxidizer. The scale on this display is common to both variables and is calibrated in terms of feet-per-second of velocity change capability. This assumes a constant specific impulse and further requires that vehicle mass be known. It appears that the principal change in vehicle mass will be due to expenditure of rocket propellants; hence, the mass could reasonably be made a function of rocket propellant remaining, thus making this scale calibration possible. The display of rocket power capability in terms of ft/sec is important since these are the units in which the computer will read out required rocket performance. The thrust display shows individual indications of both actual and commanded thrust for each of the main engines.

The remaining controls for the rocket system are on the console within the commander's cocoon. The control system shown is based upon the assumption that sufficiently accurate thrust application is not within the capability of the crew without providing a display system of such complexity as to require the same sensing components which are needed for an automatic thrust control system. On these grounds it is felt that the task can be done better with the man serving as an alternate input source to the thrust control system to provide backup for the computer. This presumes that the thrust control system can be built to operate independently of the computer and with sufficient reliability to assure mission success. This is not an unreasonable assumption since the system requires little computation and is composed principally of an inertial measurement system and a servo system. The controls on the console allow the commander to choose either manual input or computer input to the thrust control system. In the case of manual inputs, provisions are made for input of azimuth angle, elevation angle, magnitude, and time. If computer input is selected, the values of these parameters will be shown on the computer readout display (described elsewhere). Rocket operation is initiated by: pressurizing and opening valves on desired tanks, selecting thrust level (2000 lb. or 6000 lb.) on desired engine, arming the desired low-thrust engine, orienting the thrust chamber of the selected rocket through the commander's side stick and mode selector (chamber position is shown on a display to the left of the attitude indicator), actuating ignition system, and when the preset firing time approaches, actuating the normally open arming switch on the commander's left grip. Operation may be monitored by observing velocity buildup on the computer readout and by comparing commanded and actual thrust on the thrust display. If automatic cutoff does not occur or if attitude is not maintained — or for any other reason — the rocket may be cutoff by releasing the normally-open arming switch on the left grip. For operation with the cocoon closed, an emergency pressurizing control is provided on the console which will pressurize and open feed lines on all tanks. There would be some added propellant lost with this procedure, but it allows rocket operation to be accomplished without opening the cocoon.

Additional displays of attitude control system propellant quantities and pressure, along with pressurizing controls, are located to the left of the computer readout display. Required circuit breakers for the rocket engine systems are located overhead on the power distribution panel.

2.4.2.2 COMMUNICATIONS

The communication controls Figure II-2-6 are located on the left side of the commander's lower panel and on the power distribution panel overhead. All off-on controls are on the overhead panel and the mode select switching is on the lower panel. Each piece of equipment has a three-position circuit breaker which allows it to be turned off, or to be connected to either buss 1 or buss 2. Each transmitter system, VHF and UHF, has a mode select including: normal, voice/telemetry, emergency voice, and ranging. In the normal mode, two transmitters are operated for telemetry and voice transmission respectively.

In the event of transmitter failure, voice transmission is given priority and would utilize the remaining transmitter. The ranging mode allows the transmitters to be used to measure distance. An identical system is used for both VHF and UHF, the

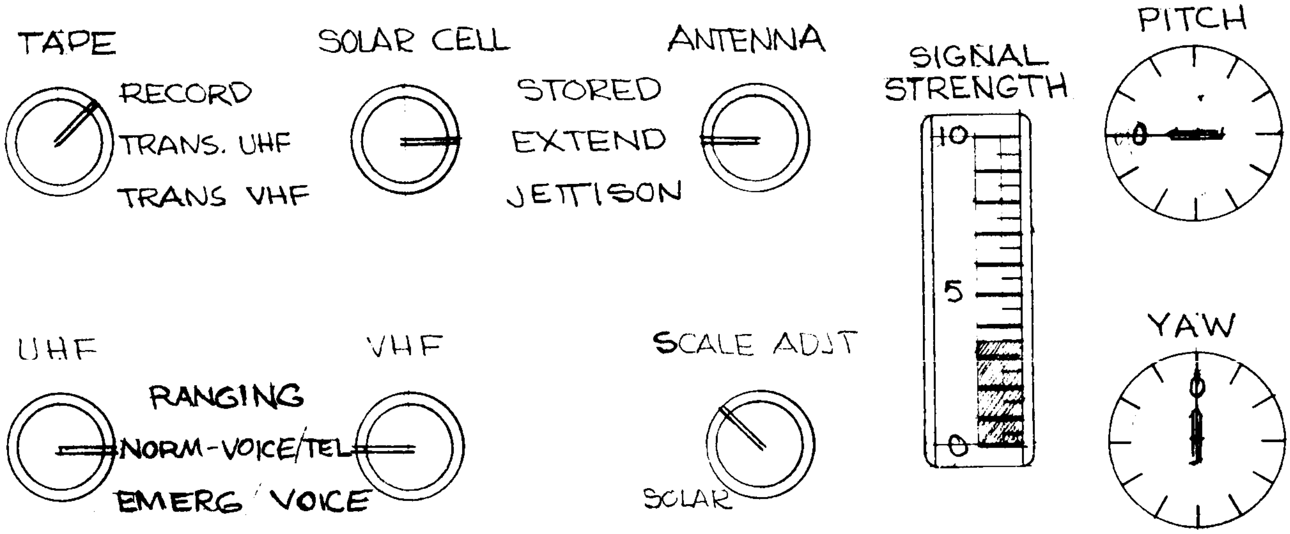


Figure II-2-6. Communications

two systems being provided to allow both near-earth and deep-space communications. For the latter, a high gain directional antenna is provided with a gimbaled mount. The crew must monitor signal strength on the meter provided and re-orient the antenna as required to retain the maximum signal. The signal strength meter has a continuously variable gain control and is calibrated in relative units since there is no need to know actual magnitude but merely to peak the signal. The gain control allows the signal strength indication to be adjusted to obtain sufficient display motion for optimum peaking even though the absolute volume at signal strength varies over a large range. The position of the antenna is indicated on the displays to the right of the signal strength indication. The antenna position may be varied by using the commander's side stick in conjunction with its mode selector. An additional control is provided to allow "extension" of the antenna after launch and its "jettison" prior to re-entry.

A data link receiver is included to allow ground-supplied data to be inserted into the computer for subsequent readout and evaluation by the crew. The digital processor encodes a number of measured variables for telemetry to the earth. A tape recorder is provided to record the output of the digital processor while behind the Moon when earth communication is not possible. The tape recorder has a mode selector to allow: record, transmit VHF, or transmit UHF. An emergency VHF voice receiver and code transmitter are provided as backup communication these are activated by switches on the power distribution panel. This equipment would operate without the benefit of the directional antenna. A code key would be stowed in each cocoon.

An intercom system is included at all stations for use during high noise condition or when the cocoons are closed. A station would also be included in the mission module.

2.4.2.3 SECONDARY POWER

Directly beneath the power distribution panel on the commander's left are the secondary power displays, Figure II-2-7. There are displays of voltage, current, pressure, and pressure rate of hydrogen and oxygen in the fuel cells. A selector switch is provided to monitor any one of the eight fuel cell batteries. The hydrogen and oxygen pressure displays are combined and shown with a time pointer. Since the hydrogen

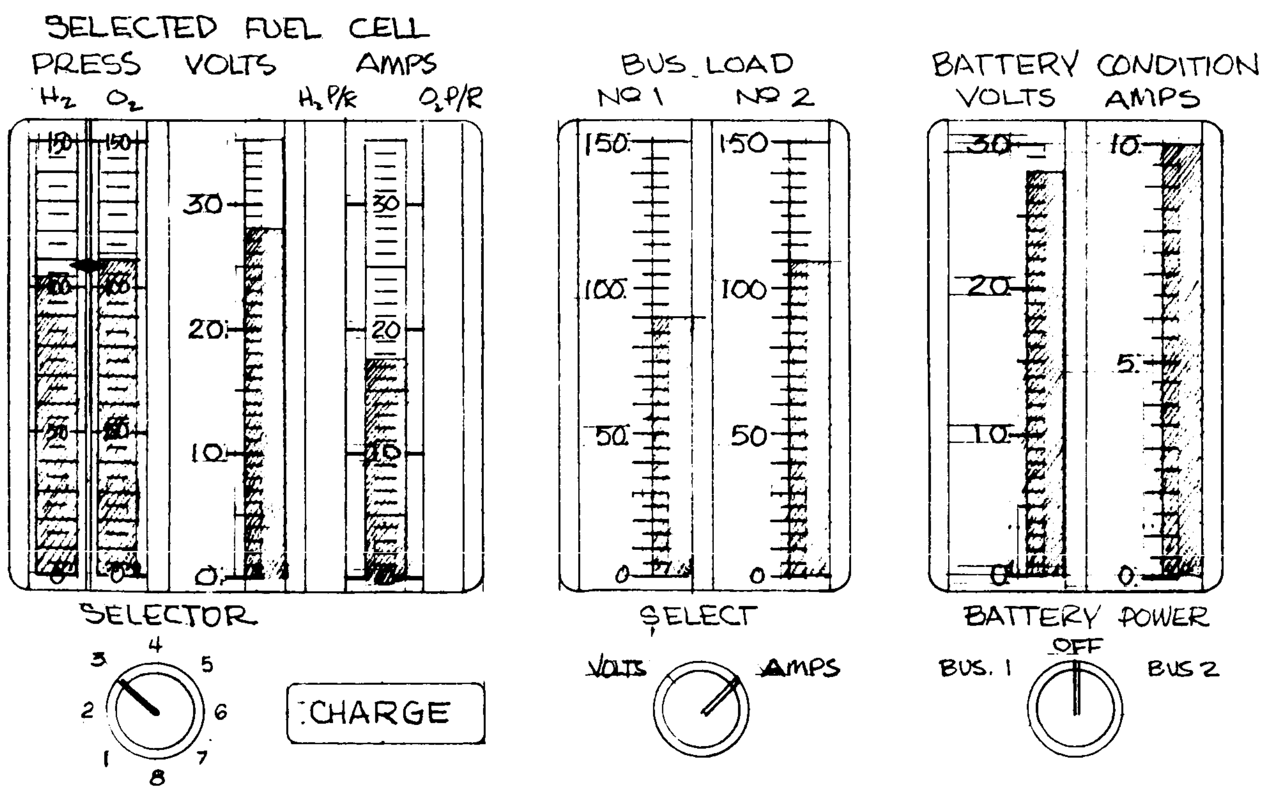


Figure II-2-7. Secondary power

and oxygen pressure are proportional to the power stored in the fuel cell battery, these indications should increase and decrease as a function of time during the charge and discharge cycles of operation. If the charge rate is either too low or too high, the pressure indications will fall below the time based reference. The time reference would move from maximum pressure indication towards zero pressure during the dark cycle of operation. During the light cycle the time reference would drive back toward the maximum pressure. This process would be repeated during each lunar orbit to provide a standard against which to evaluate fuel cell battery power. If the power display indicates a low charge rate, the solar collector position may be monitored by checking the output of a series of test cells on the signal strength meter directly beneath the secondary power displays. This meter is normally used to read antenna signal strength; however, a mode selector allows it to be used to monitor the solar cells. The charge or discharge current of the selected fuel cell battery is shown on a scale along with the pressure rate of both hydrogen and oxygen.

The pressure rate indications are calibrated so as to exactly equal the current indication for normal fuel cell operation. If an internal fire (within the fuel cell) or an external leak develops, the pressure rate indication will differ from the current indication. If this difference exceeds a preset level, the master warning light and the light in the circuit breaker controlling the related fuel cell battery will be illuminated.

A continuous indication of current in each buss in shown, and activation of a spring-loaded switch allows the reading to be changed to buss voltage. The emergency battery has a continuous display of voltage and current, as well as a switch to allow it to be connected to either buss. In the event of a complete power failure, this battery would automatically energize an ambient light, the cabin gas system, and a cabin "minimum air circulation" blower.

This should sustain life in the event the crew is temporarily disabled. The crew would then clear one of the main busses of all equipment, including fuel cell batteries by means of the power distribution panel, and then connect the battery to that buss to allow operation of selected items of equipment.

2.4.2.4 POWER DISTRIBUTION PANEL

The commander's overhead panel contains the power distribution switching and the fire detection system display, Figure II-2-8. Integral with the power switches are individual annunciators which operate in conjunction with the master caution and warning system.

Arranged in the leftmost vertical row on the panel are individual controls for each fuel cell battery. These multi-function switches allow the output of each battery to be turned off, connected to buss number 1, or connected to buss number 2. The remaining six rows contain switches, grouped according to subsystem, which provide identical switching function for each individual piece of electrical equipment.

When a switch is moved to the buss 1 position a characteristic color is displayed which differs from the color appearing in the buss 2 position. This facilitates recognition of switch position and provides a link to the buss-voltage and buss-current displays which are located on a panel below.

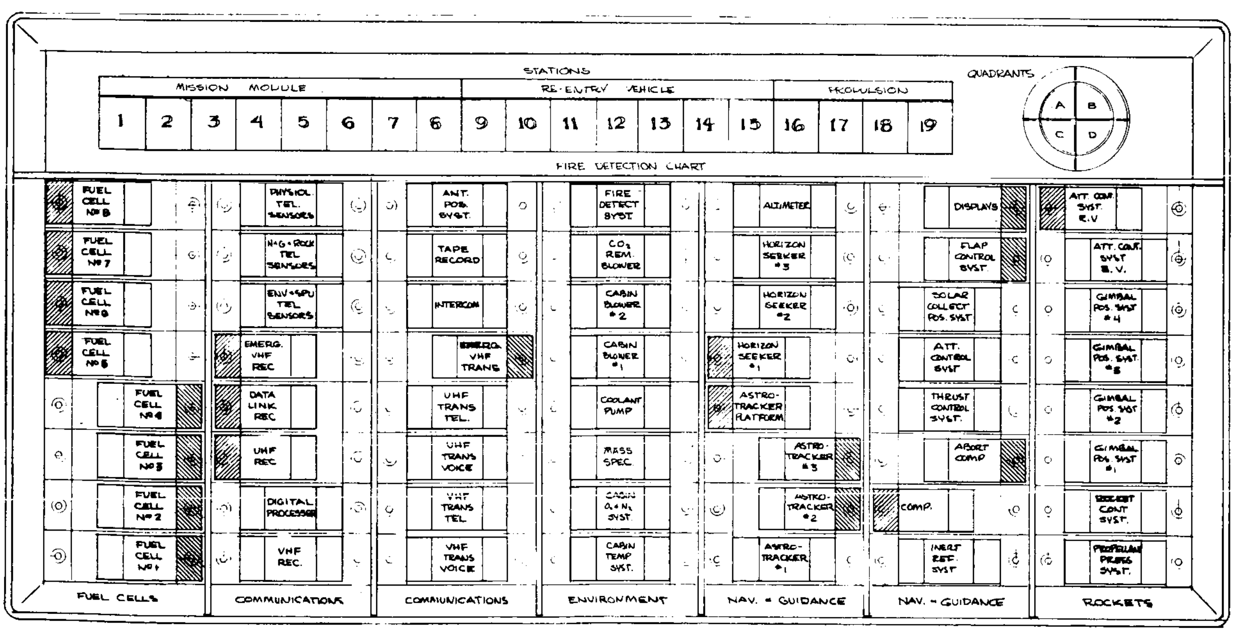


Figure II-2-8. Power distribution panel

Current overload protection is also provided by these switches. They automatically return to the center "off" position if the preset current is exceeded. The translucent portion of each switch may be illuminated by either a white or an amber light. The white light series operates in a check list capacity and is used in conjunction with the sequence display located on the vehicle engineer's panel. When a particular mode is selected on the sequence display, for example "re-entry", all the power switches which are not in the correct position will be illuminated with the white light. The crew may then actuate those switches which are illuminated without the need for checking the positions of all 56 switches.

The amber light operates in a dual capacity. It may flash to indicate a fire in a particular battery or subsystem, or it may be illuminated (nonflashing) to indicate a failure detected by the master caution and warning system. In either case, attention will be directed to the switch panel by an illuminated master caution or warning light centrally located on the main panel. In addition an audible signal will be sounded as a fire warning. Realizing that many of the subsystems will have components widely scattered throughout the vehicle, a further warning, serving to localize the fire, is provided by the station and quadrant readout above the power distribution panel. The vehicle is divided into a number of stations along its longitudinal axis and into quadrants about that axis. The quadrants are further subdivided into pressurized and unpressurized areas. This station and quadrant display will allow the source-of-fire warning to be quickly located and thus expedite any possible emergency action.

2.4.2.5 ENVIRONMENT PANEL

This panel Figure II-2-9 provides continuous displays of principal environment parameters. It also has provision to permit interrogation for any of twenty less important parameters as an aid to diagnosing system malfunction. The displays in the upper left corner show oxygen and nitrogen quantities for each individual tank as well as a total for each gas. The displays of total oxygen and nitrogen quantity are on a single indicator which also has an indication of total elapsed mission time. If either quantity falls below the elapsed time indication, the rate of use must be curtailed or the mission will have to be shortened to avoid depletion of the expendable involved. The individual quantity displays allow the crew to adjust the system so as to maintain approximately equal quantities in each tank, thus avoiding the possibility of developing a leak

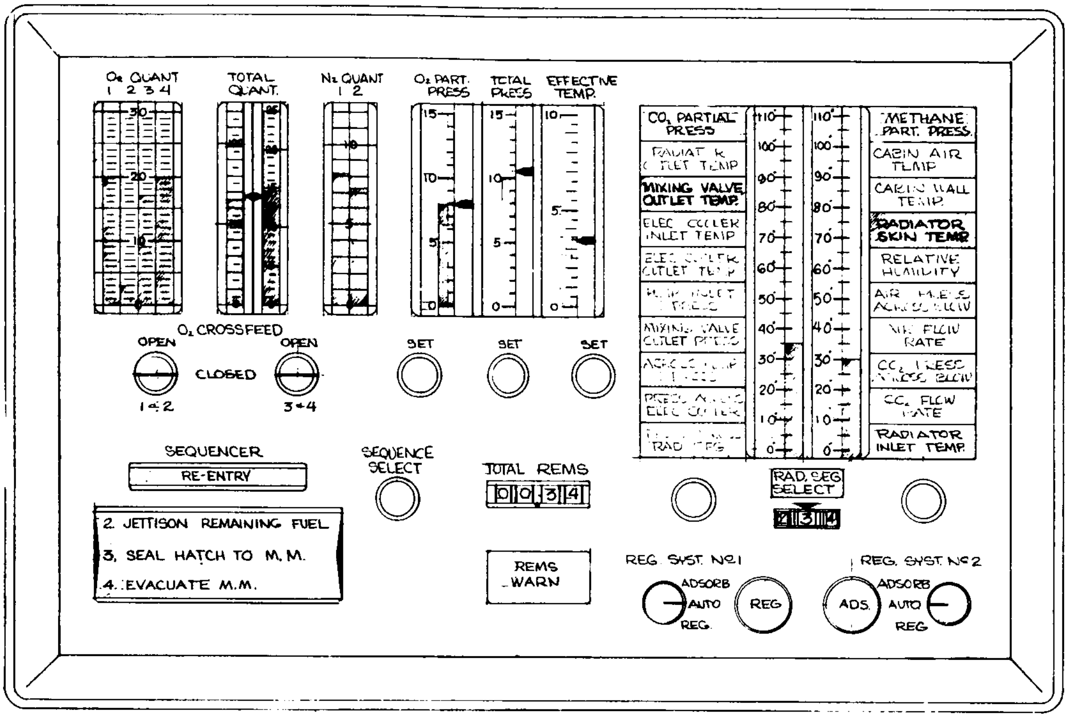


Figure II-2-9. Environment panel

in a full tank at a time when the other tanks are nearly empty. No valve controls are shown for the environment storage tanks since they are to be within the re-entry vehicle. These valves would be operated directly by the crew.

Beneath the quantity displays are two crossfeed valve controls. By pressurizing the rocket oxygen tanks and operating the crossfeed valves, oxygen could be transferred from the rocket system to the environment system.

To the right of the quantity displays are located the cabin-environment controls and displays. Oxygen partial pressure and total pressure, with their associated set knobs, are self-explanatory. The effective-temperature readout and corresponding set knob are calibrated in abstract terms since effective temperature is a function of several factors such as cabin air temperature humidity, airflow, etc. Thus, for a single setting of the control, air temperature may vary with humidity, making absolute scale markings impractical.

On the upper right of the panel is the interrogation readout. Two scales are available, each with ten selective parameters. Selection is accomplished with the rotary switches beneath the displays. As each of the various parameters is selected, the corresponding name will be illuminated, in the left column for one display and in the right column for the other display. The scale markings on the readout will change to provide the correct range in the correct units.

Directly beneath these displays are the mode displays for the two CO_2 removal systems. Override switching is also provided to allow manual cycling in the event the automatic sequencing fails.

A cumulative radiation readout and a warning light for high radiation rate are provided. In conjunction with the radiation rate warning, CO_2 buildup, and low oxygen partial pressure, a distinctive audible warning signal will be sounded.

The presence of toxic material in the cabin atmosphere will be detected by the mass spectrometer. When out of tolerance conditions are detected, this condition will be signalled by a blinking light located on the life support system interrogation-readout panel. In addition the mass spectrometer switch on the power distribution panel will be lit from behind when the device is out of calibration.

2.4.7.6 SEQUENCING DISPLAY

On the lower left is the sequencing display with its associated controls. This provides a convenient storage device for check-list information. It is also tied into the power distribution panel as previously described. The events contained on the check lists are those events which are accomplished at a time determined by a complex function of several variables and hence, cannot be easily programmed for automatic operation. The events must, however, be accomplished in the correct sequence. For those events which are pre-programmed as a function of time and over which the crew has no control, a voice presentation will be used. For example, during launch the numerous staging events would be called out and verbal warning would be given if an event failed to occur. This would provide some anticipation to the commander of the impending abort.

2.4.2.7 VIEWER

Centrally located on the commander's console is an optical viewer with several modes of operation, Figure II-2-10. In the Viewing mode, the optics provide a choice of wide angle or normal lenses and several filter colors and densities. Integral with the external view are reticle lines which may be adjusted so as to just frame either the earth or the Moon when viewed through the wide angle lens. By choosing the correct reference (earth-Moon) an approximation of altitude may be read off a counter attached to the recticle position system. The wide angle lens also allows rough attitude information to be derived from the viewer by noting the position of the earth or Moon image in the field of view. In the orientation mode, the same external view is presented with the addition of a superimposed predetermined matrix of stars. By orienting the vehicle manually so as to superimpose the view of the actual stars with the projected matrix, the attitude of the vehicle can be re-established in the event the platform loses its reference. A "sychronize" — "normal" mode switch is provided to allow the platform to be sychronized to the proper position. At the time the star images coincide with the matrix, the platform would be switched to normal.

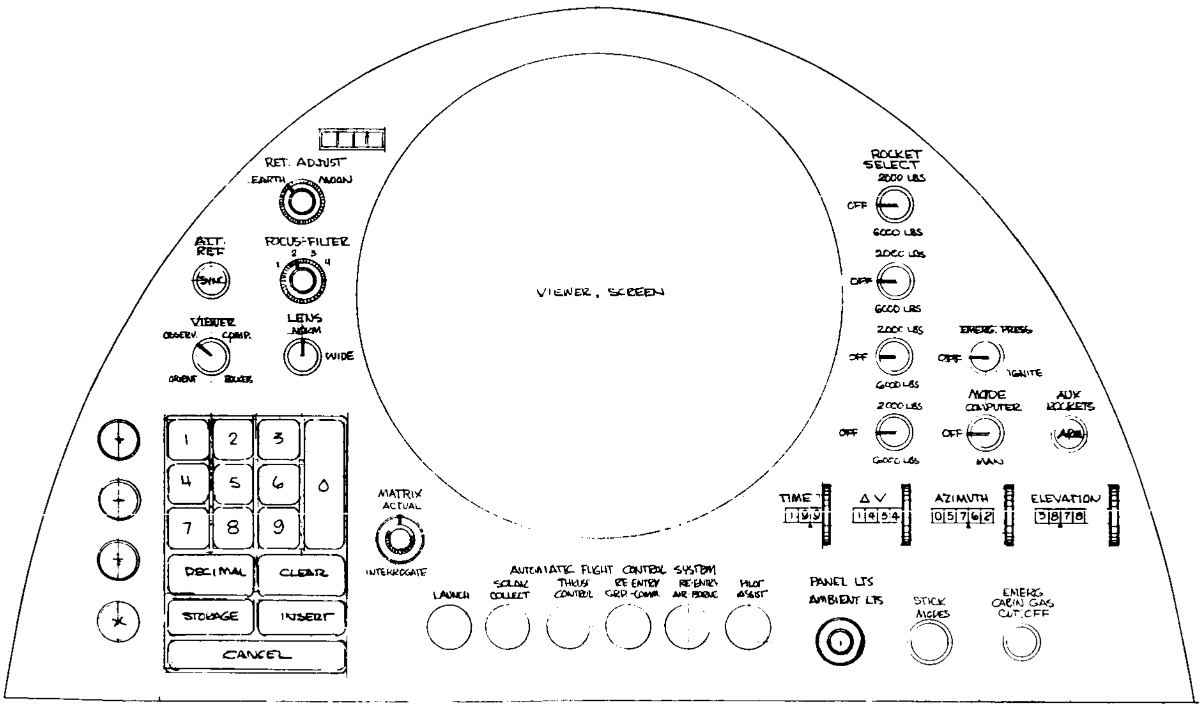


Figure II-2-10. Viewer

A third mode, "computer", allows a redundant composite readout cathode ray tube to be displayed through the viewer.

In the "re-entry" mode the viewer would have that angle of view and that magnification which would allow the operator to see the portion of the earth toward which he is heading in the late stages of re-entry. He would use this information first in aerodynamic control and later in manipulating the parachute to avoid local obstacles and connect for lateral velocities.

To allow maximum use to be made of the computer, a flexible data insertion and readout capability has been provided, utilizing a key board, a mode selector, and a cathode ray tube. The mode selector allows any one of several predetermined display matrices to be displayed on the cathode ray tube, Figure II-2-11.

A typical matrix is shown below:

LUNAR AIM POINT

MISS DISTANCE

X ____________________ 1
Y ____________________ 2
Z ____________________ 3
Time ________________ 4

VELOCITY INCREMENT

Magnitude ____________ 5
Azimuth ____________ 6
Elevation ____________ 7
Time ________________ 8

VELOCITY, TOTAL

Magnitude ____________ 9
Azimuth ______________ 10
Elevation _____________ 11
Time ________________ 12

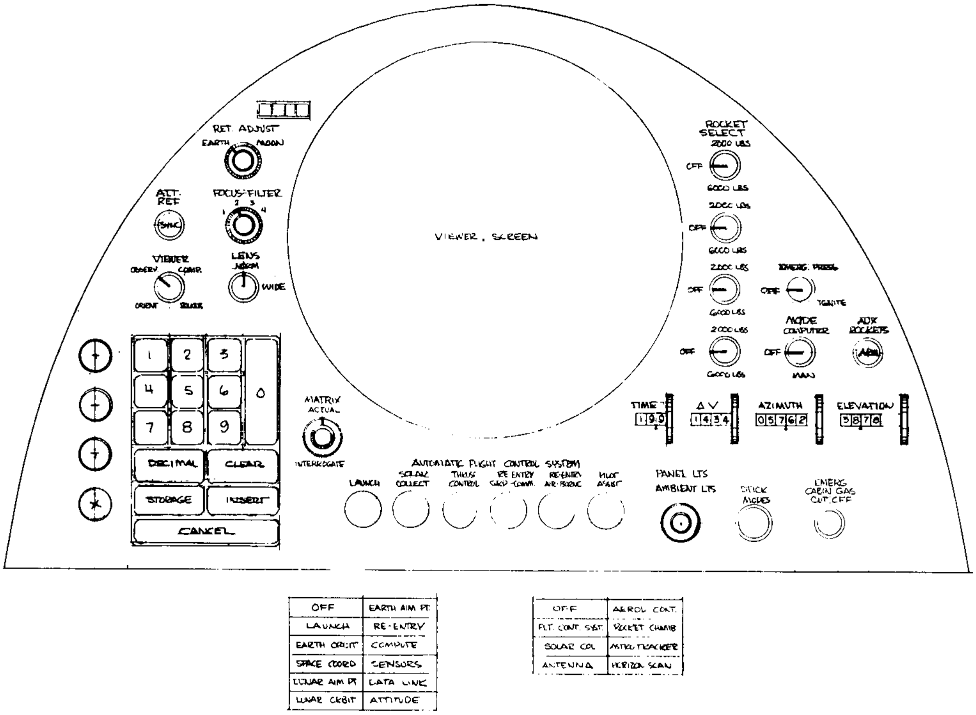


Figure II-2-11. Computer readout

Each of these parameters would appear in a numbered space on the display. The mode selector also provides for a selection of actual, or interrogated, values of those parameters. If "actual" is selected, the readout of appropriate values will occur immediately. If "interrogate" is selected, no parameters will be read out until sufficient initial conditions have been inserted to define the problem. Inserted data will be differentiated from readout data by means of color. Data insertion is accomplished by first punching the number of the desired parameter on the keyboard and depressing the address button. This directs further inputs to the proper address within the display matrix selected. The desired value of the selected parameter is punched in and appears on the readout for checking. If the correct number is displayed it is then inserted into the problem by means of the insert button. When sufficient inputs have been provided to allow a computation to be made, such will automatically occur. Provisions for decimals, clearing previous mode selections, and cancelling inserted data are available. This type of display and insertion system allows a great deal of information transfer without burdening the system with an extremely large, difficult-to-interpret mode selector.

Further study will be required to completely define the number and exact configuration of display matrices required to allow optimum interchange of information. The following is a list of possible modes with a brief statement of the type of information each might display.

LAUNCH	- Data to monitor launch and determine abort procedures.
EARTH ORBIT	- Standard orbital parameters plus velocity increment parameters for evaluating the effect of thrust application.
SPACE COORDINATES	- Position and velocity in space coordinates plus velocity increment parameters.
LUNAR AIM POINT	- Miss distances, velocity increment parameters, and total velocity.
LUNAR ORBIT	- Same as earth orbit, but based on a Moon-centered coordinate system.
EARTH AIM POINT	- Similar to lunar aim point, plus re-entry parameters.
RE-ENTRY	- Plot of attainable landing area with relative positions of desired and predicted landing sites.
SENSORS	- Provision to monitor inputs from astro trackers, horizon scanners, etc.
DATA LINK	- Ground command data
ATTITUDE	- Inertial measurement unit outputs.
COMPUTER	- A blank matrix for insertion and manipulation of numbers.

2.2.4.8 SIDE STICK CONTROLLER

A three axis controller is mounted on the right armrest inside the commander's cocoon. A mode selector for the stick is located on the commander's console. Modes are provided to allow movement of the antenna and the rocket chambers. When one of these modes is selected, the position of the device being controlled is shown on a time-shared display to the left of the attitude indicator. Astrotracker and horizon scanner pointing capability is also provided. In either of these modes the position of the sensor being pointed must be monitored on the computer readout display. The desired astrotracker or horizon scanner is selected by selecting the "point" mode by means of the mode switcher located to the left of the computer readout cathode ray tube. When a signal sufficient for lock-on is obtained, it will light the "lock-on mode". At this time the lock-on may be selected for that sensor. A similar mode selector is provided for the platform stellar reference. If maneuvering limits are exceeded and the reference looses its stellar monitor, the monitor will automatically go into the "synch" mode. At that time the vehicle must be re-oriented to a predetermined attitude and the reference monitor locked on again. Orientation of the vehicle may be done several ways. The side stick may be used directly to control the aerodynamic controls during atmospheric flight. During space flight the "flight control mode" may be selected which then allows the attitude to be controlled through the reaction jet control system. In this mode, control action will command angular rates about each of the vehicle axes.

Other automatic flight control modes are available and may be selected by means of switches on the commander's console. In order to anticipate the control action which will occur when a mode is selected, it may first be displayed on the command needles. This also allows the operation of the flight control system to be monitored after engagement, by observing the command needles which should remain centered.

2.4.2.9 LEFT ARMREST CONTROLS

The commander's left armrest, Figure II-2-12, contains three controls, an abort switch, a secondary mission switch, and a safety switch. The first two are self-explanatory and the latter is provided as a safety interlock for all irreversible switching operations. For example, before the antenna, solar collector, or mission module can be jettisoned, this switch must be actuated as well as the mode switch for the item being jettisoned. This guards against accidental operation of an important switch while entering or leaving the seat. The switch is also interlocked with the rocket engines and must be held closed any time the rockets are fired.

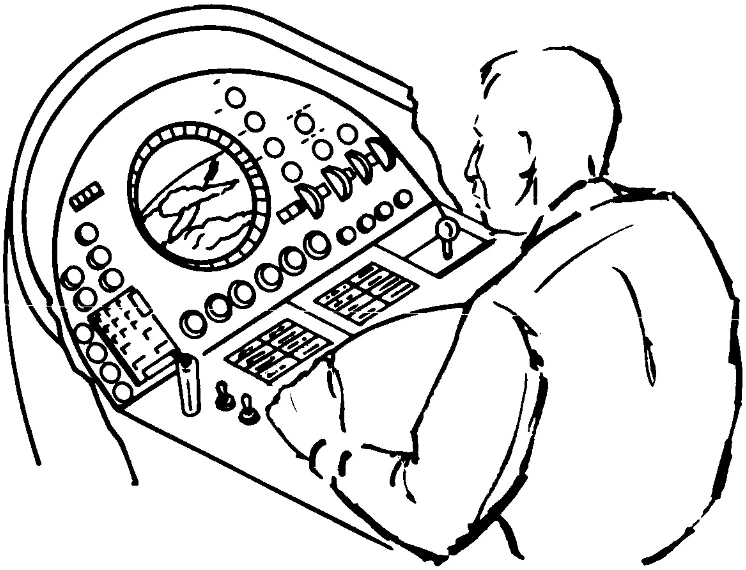


Figure II-2-12. Armrest control

2.4.2.10 CHECKLISTS

The information contained on the following check lists are tasks required by the operator in performing some of his control functions within the system. It should be noted that the tasks are associated with abort, midcourse navigation, lunar orbit, earth orbit, and attitude control during trajectory corrections and are viewed as discrete rather than of a continuous nature. The principal function where this is not the case is attitude control during periods of relative light control inactivity, i.e., when no changes in the trajectory are being staged. This situation may be contrasted to the continuous control function of the operator in the manual mode of re-entry flight control.

When control tasks of the pilot are discrete, they may be adequately described in terms of check lists. In effect, these check lists provide a description of the contemplated control tasks required of the operator.

Some preliminary checklists are as follows;

2.4.2.10.1 Velocity Correction

1. Select "VELOCITY CORR" on sequence display
2. Check power distribution panel observing buss load displays
3. Select "COMPUTER" or "MANUAL" input to thrust control system If "MANUAL" –
   a. Select "DATA LINK" on computer readout or obtain parameters by voice communication
   b. Set magnitude
   c. Set azimuth angle
   d. Set elevation angle
   e. Set time to fire
   f. Select "ROCKET FIRE" on time-to-go mode selector
4. Check thrust level on engine desired
   a. Select "Thrust Chamber" mode on commander's side stick
   b. Observe chamber position on display
   c. Manipulate stick to obtain nominal position
5. Alert crew to take normal duty stations
6. Select "Thrust Control" on command needle mode selector
7. Select "Thrust Control" on flight control mode selector
8. Observe command needles, attitude display, vehicle rate display
9. Pressurize desired fuel and oxidizer tanks
10. Open feed valves from pressurized tanks
11. Select "Lunar Aim Point" on computer readout

12. When time-to-go approaches zero, turn on ignition system

13. At 15 seconds to go, actuate normally open safety switch on commander's left armrest.

14. As rocket fires, observe:

    a. Thrust

    b. Total velocity on computer readout

    c. Command Needles on attitude indicator

    d. Body Axis Rate Display

15. Release safety switch if automatic cutoff fails or if for any reason premature cutoff is desired.

16. Re-orient Solar Collector and Antenna

2.4.2.10.2 Solar Collector

1. Extend Solar Collector

2. Select "Solar Cell" on command needles

3. Select "Solar Cell" on flight control system

4. Observe vehicle re-orientation

5. Check solar cell output on signal strength meter

6. Check charge rate of individual fuel cell batteries

2.4.2.10.3 Antenna

1. Extend antenna

2. Select "Antenna" on commander's side stick

3. Observe antenna position and signal strength display

4. Rotate antenna about one axis to achieve maximum signal strength

5. Adjust gain of signal strength display as desired

6. Rotate antenna about other axis to achieve maximum signal strength
7. Repeat about first axis.

2.4.2.10.4 Re-Entry

1. Select re-entry on sequence display and computer readout
2. Check power distribution panel
3. Jettison solar collector
4. Jettison antenna
5. Crossfeed excess oxygen from rocket system to environment
6. Seal hatch to mission module
7. Purge mission module
8. Check hatch for leaks
9. Turn on re-entry vehicle attitude control system
10. Jettison mission and propulsion module
11. Re-align vehicle with velocity vector
12. Select desired mode on command needles
13. Select desired mode on flight control
14. Observe re-entry displays and monitor system generation
15. Change mode of operation or fly manually if required
16. Verify parachute deployment through periscope
17. Actuate mechanical backup controls if parachute fails to deploy or impact bag does not inflate
18. Observe periscope to avoid obstacles
19. Verify operation of rescue aids.

2.4.2.10.5 Lunar Orbit

1. Select "Lunar Orbit" on sequence display
2. Select "Lunar Orbit" on computer readout
3. Check power distribution panel
4. Check secondary power system
5. Evaluate fuel requirements to achieve desired orbit
6. If feasible, initiate velocity change to achieve orbit
7. Observe Moon through periscope
8. Verify altitude readings with periscope
9. Monitor antenna signal strength to maintain communications
10. Turn off telemetry when behind Moon
11. Turn on tape recorder when behind Moon
12. Check operation of fuel cell batteries when transitioning between dark and light portion of orbit
13. Select "Earth Aim Point" on computer readout
14. Determine desired lunar orbit departure time
15. Initiate velocity change to intercept earth aim point

2.4.3 Re-entry Controls and Displays

2.4.3.1 CONTROL TASKS

There are several operating modes for re-entry flight control and navigation. These are: 1) a completely manual mode (damping system operative), 2) a completely automatic mode based upon commands from the flight data computer or ground-based computer for some nominal trajectory, and 3) pilot control mode based upon commands of the flight data computer or ground-based computer for some nominal trajectory.

In the complete automatic mode, the pilot must decide which source of command information he wants to fly and then switch to it. In addition, he is able to monitor the situation using the displays provided him for the complete manual mode. To make his monitoring task of the "how goes it" situation much easier, the flight director will display the commanded values of vehicle attitude.

In the pilot control mode based upon commanded signals, the pilot again must select the command source. He then tracks the director needles in flying the nominal trajectory for navigating to the pre-selected target (desired x, y, z position). Definition of the quickening function to be displayed on the director needles requires further study.

Of primary interest, is the complete manual mode of operation. The task objective of the operator in this mode is to navigate the vehicle safely to some pre-selected destination from some set of initial conditions without explicit reference to some precomputed nominal trajectory. If it is assumed that the vehicle is within the re-entry corridor (has a safe re-entry angle) and that the stability augmentation system adequately damps all short period oscillations, the operator's control task may be described as follows:

1. Maintain a trim pitch attitude angle (recovery vehicle attitude control system) and a zero roll and yaw angle when the vehicle is re-entering at escape velocity and dynamic pressure is gradually increasing from zero. The predesignated target should be within the precomputed minimum-maximum down range and lateral range capability of the vehicle at this time.

2. Maintain this flight condition until zero altitude rate is obtained. When zero altitude rate is arrived at, initiate a 180 degree roll so that negative lift may be employed in flying a constant altitude trajectory.

3. Maintain a constant altitude trajectory by controlling negative lift (variable pitch attitude) until sub-orbital velocity is obtained. It is also possible at this time to make some lateral and down range corrections so as to keep the predesignated target within the center of the precomputed range capability of the vehicle. This is accomplished by varying pitch attitude (lift) for down range corrections. The same technique is used for lateral range corrections in that only lift is used. However, the vehicle is rolled over to some desired roll angle where the lift vector is lateral to the relative wind.

4. When sub-orbital velocity is arrived at, initiate a 180 degree roll angle so that positive lift may now be employed in flying an equilibrium glide trajectory. This equilibrium glide trajectory may be approximated by maintaining a constant sinking rate or altitude rate. Adjustments should also be made to minimize target miss distance. It should be noted, however, that most of the range capability of the vehicle has been expended in the higher velocity regions of the re-entry trajectory.

5. By the time 50 percent of orbital velocity is obtained, the "G" and heating problems are no longer present. In addition, there is little remaining range capability and consequently, the vehicle should be zeroed in on its target at this point. At the approximate altitude, the drogue chute and reefed main parachutes should be released.

If the initial conditions are ideal, i.e., correct re-entry angle and target directly within center of the minimum and maximum lateral and down range capability of the vehicle, the pilot should be able to control and navigate his vehicle to the pre-selected target by following the above procedures. If the initial conditions are not ideal and the pilot must maneuver the vehicle close to the upper or lower re-entry corridor boundaries, the task will be considerably more difficult. The important consideration in defining the pilot's task under these circumstances is that he does not have continuous aerodynamic control (the control surfaces are ineffective during various portions of re-entry). In addition, the closer to the boundary limits the vehicle operates, the less there is of effective aerodynamic control with respect to time. Consequently, the pilot must be able to anticipate the result of his control actions.

When operating at the upper limit, i.e. shallowest re-entry angle, because of guidance error or attempting to fly maximum down range, the problem is one of remaining in the atmosphere, e.g., preventing skipout. The approach taken here is that this can be best accomplished by employing negative lift rather than by employing drag. The danger, of course, is that negative lift may not be employed by the pilot soon enough or long enough. For this reason, some type of safety factor should be built into the displayed computations of predicted maximum-minimum range and skip limits.

On the other hand, when operating at the lower limit, the problem is not to exceed the "G" or heating limits. In this situation, positive lift should be employed. However, it must be employed with care because a certain magnitude of "G" and heat absorbtion is necessary and desirable. Although these desirable amounts have not been defined for display purposes, it would conceivably be advantageous to incorporate desired "G" or temperature rate in the display of these parameters. This has not been done because of a possible confusion between command signals originating from flight director needles and these parameters when flying re-entry in the nominal trajectory command mode.

In developing displays for re-entry, information which reflects three different aspects of the operator's control task during re-entry were conceptualized. These task categories are: 1) long range planning, 2) immediate situation and 3) immediate vehicle orientation. A fourth category was established which provided redundancy with respect to the critical parameters having well defined limits, i.e., skin temperature, total heat absorbed, and "G". Within this general framework, an analysis of the control and feedback loops required for operator control during re-entry was possible.

The block diagram below, Figure II-2-13, illustrates these information categories, starting from the general planning parameters to the more specific attitude control parameters. In effect, the diagram defines the feedback parameters and the control modes available to the pilot during re-entry.

It should be noted that angle of attack is not a displayed parameter. It has been omitted because of the close parallel between its value and that of pitch attitude during most of the re-entry trajectory. Furthermore, as the selected vehicle configuration (D-2) uses parachutes for landing, there was no requirement for its display.

Another omission not to be overlooked is the lack of re-entry angle information. This display is considered to be part of the midcourse guidance requirements and thus, is not discussed in terms of re-entry. The midcourse guidance accuracies must be adequate in producing safe initial re-entry conditions that place the target destination within the range capability of the vehicle. In the event that these initial conditions are not safe, only fate can prevail. However, if the target destination is not within range, re-entry can still be accomplished although landing will not be at the desired location.

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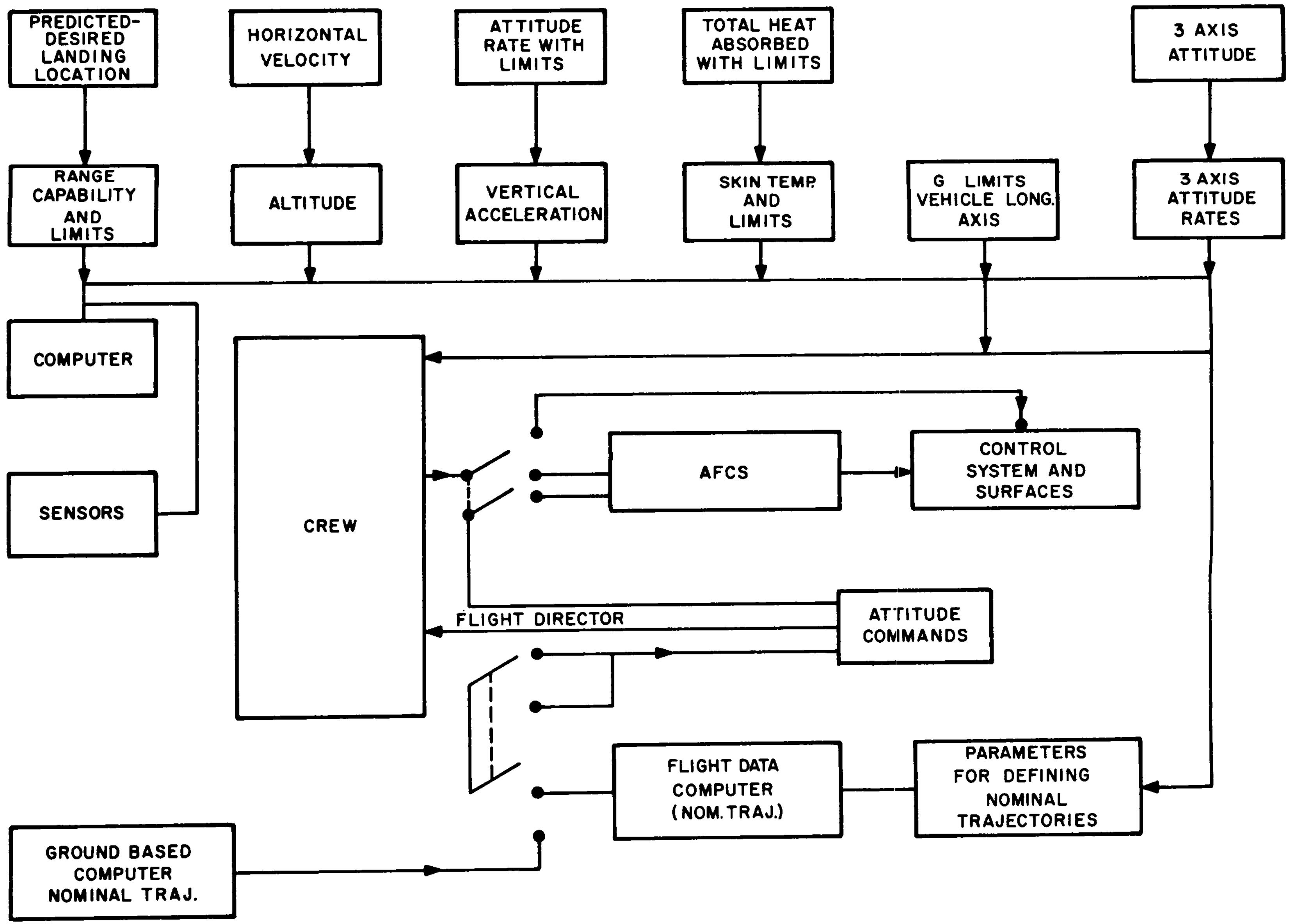

Figure II-2-13. Block diagram describing the feedback parameters and control modes available to the pilot during re-entry

[redacted]

With respect to landing preparation, chute releases will be automatic. However, control redundancy is built into the system by providing the crew with manual backup for parachute release. These controls are not located on the panel but are assigned to the space vehicle engineer's station.

2.4.3.2 RE-ENTRY PANEL AND DISPLAYS

Figure II-2-14 shows the panel location of the displays used for re-entry. The energy management, attitude indicator, and 3-axis attitude rate displays are located directly in front of the pilot-commander while the remaining re-entry displays (altitude rate, altitude, horizontal velocity, "G", total heat absorbed, and skin temperature) are located immediately adjacent and to the right.

Because of the importance of this display, the possibility of time-sharing them by locating them in a crew-shared viewing area has been examined. However, because of the effects of deceleration (limited peripheral vision) this notion was discarded in favor of locating the displays directly in front of the pilot commander where visual distortion due to deceleration is at a minimum.

The energy management display, altitude rate display, and the attitude indicator, are individually described and discussed on the following pages.

2.4.3.3 ENERGY MANAGEMENT DISPLAY

This display presents the desired destination and the predicted destination of the re-entering vehicle on a variable range scale, Figure II-2-15. The distance error between predicted and desired destination can be manually computed from the variable scale. By using a variable scale, reading accuracy may be increased as the range capability and the velocity of the vehicle are reduced. It should be noted, however, that the lateral range scale is fixed. This is because of the limited lateral range capability of the vehicle.

The displayed maximum-minimum "range bars" not only permit direct reading of the vehicle's range capability at any one time but also allow for the possibility of navigating the vehicle within the center-of-range capability. The upper range bar is driven as a function of predicted skip limits plus some safety factor and vehicle capability. The lower range bar, on the other hand, is driven by a function of predicted limits with respect to "G", total heat absorbed, and temperature plus some safety factor and vehicle capability.

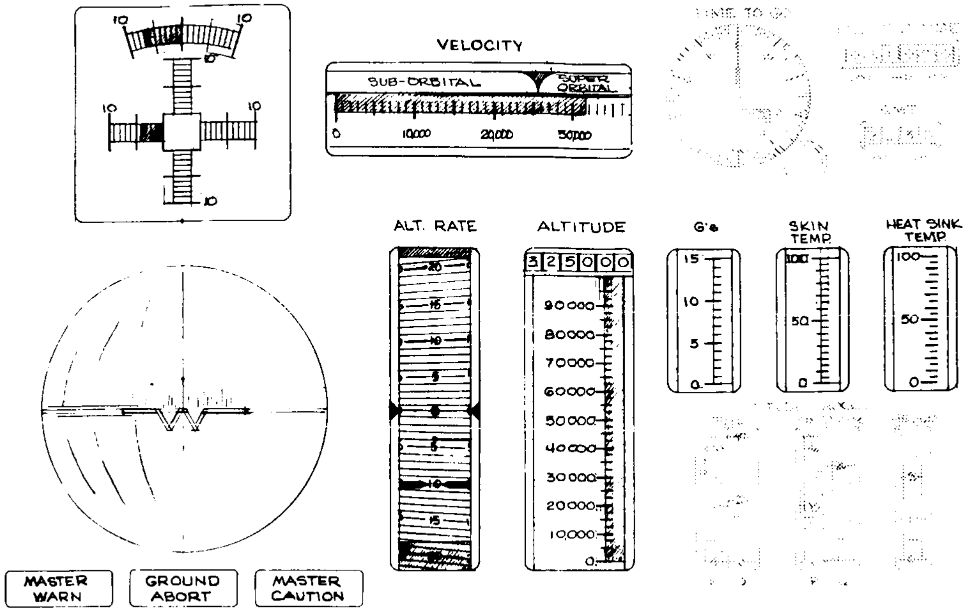

Figure II-2-14. Re-entry displays

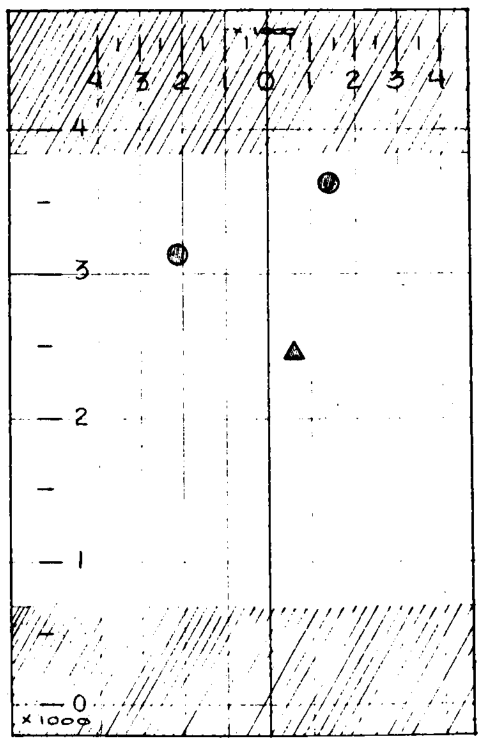

Figure II-2-15. Energy management display

In effect, this display permits the planning of a trajectory which stays within the projected corridor boundaries and range capability of the vehicle. At the same time position error is obtained with various degrees of accuracy.

2.4.3.4 ALTITUDE RATE DISPLAYS

The altitude rate scale, Figure II-2-16, is designed to give present situation information in relation to the re-entry boundary limits. In addition, "quickened" information is provided the pilot to help him avoid these limits.

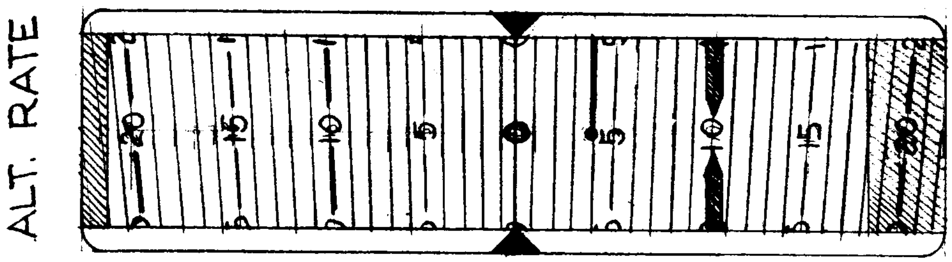

Figure II-2-16. Altitude rate display

This display is the principal input to the pilot indicating when it is time to roll 180 degree and to apply negative lift e.g., when altitude rate changes to zero during the initial re-entry run. Moreover, it will provide adequate feedback for the tasks which require the holding of a constant altitude during the super-orbital phase of re-entry and a constant altitude rate during the suborbital phase. The acceleration dot will facilitate these tasks.

The altitude rate limits are conceived to be actual rather than predicted limits. Predicted limits in the form of range limitations are incorporated in the Energy Management Display. Thus, the pilot is always aware, in terms of altitude rate, of where he is in relation to his upper and lower re-entry corridor limits. The "quickened" display permits anticipation of direction or deviations from a constant altitude or constant altitude rate flight condition.

The information is presented on a variable scale (moving horizontal scale) in order to provide varying degrees of sensitivity during different phases of re-entry. This scale is driven as a function of horizontal velocity so that the scale interval is expanded during the portions of re-entry where horizontal velocity is large. The scale interval is contracted, although the visible scale range is increased, during the portions of re-entry where horizontal velocity is relatively small and the vertical component is large.

2.4.3.5 ATTITUDE INDICATOR AND RATE DISPLAYS

Because of the attitude and attitude rate displays are the principal vehicle orientation displays, they serve a general function in all mission phases. Of particular importance are the attitude rate displays for controlling vehicle attitude during the midcourse guidance and orbital phases of the mission, Figure II-2-17. They greatly aid pilot performance when the task is to null various attitude rates or to hold a particular vehicle attitude.

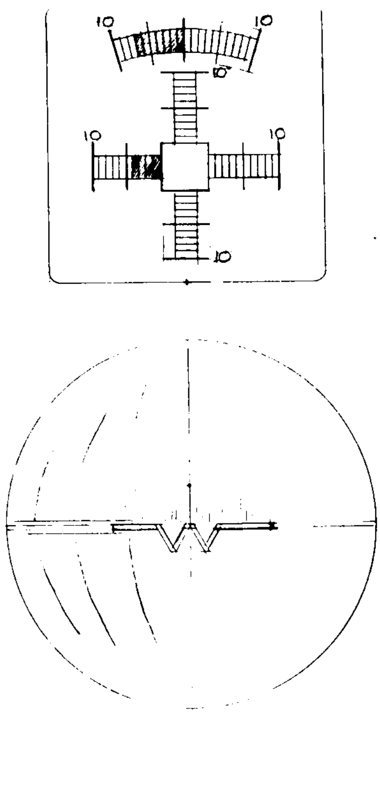

Figure II-2-17. Attitude indicator and rate displays

The 3-axis attitude indicator, of course, provides the pilot with information of the present vehicle orientation during all the mission phases. Commanded attitude information required during launch and thrust correction maneuvers are provided by the director needles on the display.

During re-entry the 3-axis attitude rate displays provide feedback to the pilot with respect to the build-up in dynamic pressure as it affects vehicle response. When dynamic pressure is small, the aerodynamic response of the vehicle is sluggish. However, as dynamic pressure increases, vehicle response becomes more dynamic. This display will indicate these different response rates and, consequently, aid the pilot in controlling the variable lift trajectory of the vehicle.

The 3-axis moving horizon attitude indicator was selected over other attitude displays because it permits the display of interpretable attitude information in all axes. For example, during re-entry the pilot must be able to control pitch attitude in flying a variable negative lift trajectory when rolled 180 degree. This upside down position is a problem in control-display design because in this attitude the usual flight control-display relationships are reversed. Rather than design two separate display systems, one for 0 degree roll angle and the other for 180 degree roll angle in attempting to provide control display compatibility, it was considered advisable to show this situation through an attitude display. It is believed that the 3-axis moving horizon attitude indicator adequately provides the operator with this information, i.e., pitch attitude and heading orientation throughout all roll angles.

It should be noted that the director needles on the attitude indicator will permit the pilot to monitor the command attitude signals originating from either the flight data computer or ground computer when re-entering in the automatic mode. A variation of this mode permits the pilot to control the vehicle while tracking these commanded attitude inputs. However, this latter mode of operation is seen as the least desirable of all the re-entry modes because servomechanisms are more capable in performing this task than the pilot.

2.4.3.6 ALTITUDE AND HORIZONTAL VELOCITY DISPLAYS

The moving pointer horizontal velocity scale, Figure II-2-18 is primarily used to indicate when the vehicle passes from super-orbital velocity to suborbital velocity, so that the pilot may anticipate when to re-initiate the 180 degree roll angle and commence an equilibrium glide trajectory. In addition, he is able to obtain some indication of his gradually decreasing range capability, and indications as to when problems such as heating, high G levels, and skipping out of the atmosphere are definitely over.

The moving pointer, digital readout altitude indicator provides both gross and refined altitude information. This information is considered as planning data for the z-axis while the energy management display is considered as a planning display for the x- and y-axis of an earth reference system. It will also provide the pilot with an indication of remaining range capability of the vehicle.

Itemized on the scale are altitudes for drouge chute release, release of reefed main parachutes, and un-reefing of main parachutes.

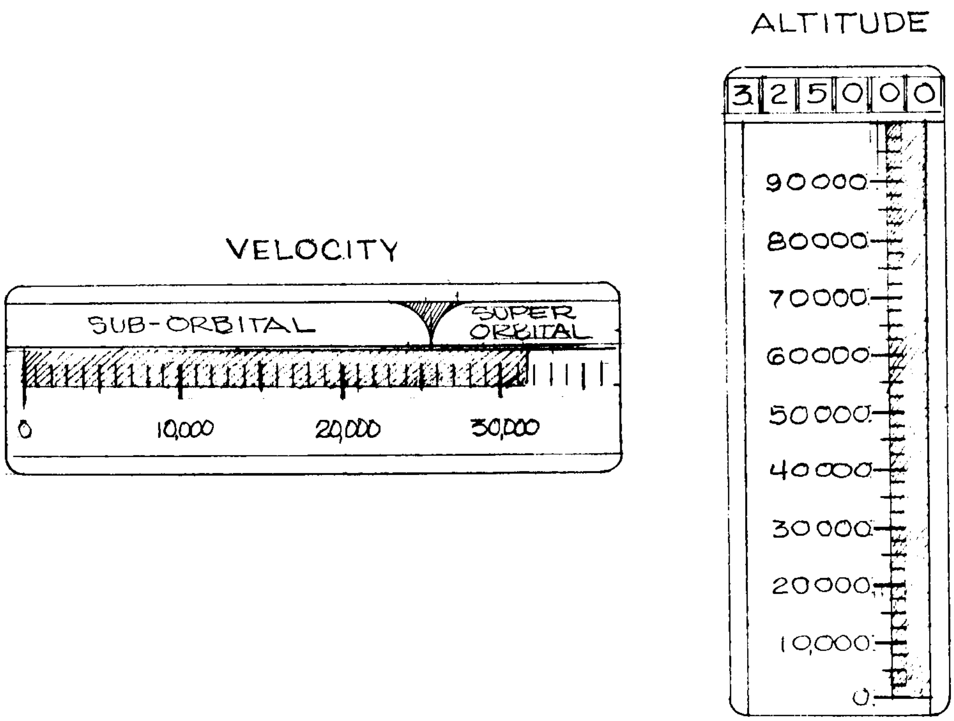

Figure II-2-18. Altitude and horizontal velocity displays

2.4.3.7 G, TOTAL HEAT ABSORBED AND SKIN TEMPERATURE DISPLAYS

The G, total heat absorbed and skin temperature displays, Figure II-2-14 are used to call out the exact values for these constraint parameters and to show, for the individual parameters, the relationship between limits and present condition. These displays are essentially backup displays to the predicted range limits on the Energy Management Display and the computed limits on the Altitude Rate Display. However, these are considered necessary information requirements in the attempt to provide the pilot with complete information about his present status. In this context, skip limits would be another required display. However, this parameter, as yet, is not as clearly defined as those above and, hence, it is only indicated among the computed items.

2.4.3.8 GLIDE RE-ENTRY LANDING (FOR THE R-3 RE-ENTRY VEHICLE)

In order to permit maneuvers below orbital velocity, glide vehicles capable of re-entry and landing at a pre-selected landing site would require additional displays besides those which have been described.

To provide an indication of the lift-drag ratio, the stall limits and the proper flare out of the vehicle, angle of attack information is required. Angle of attack being indicative of the lift-drag ratio, is used to define the maximum and minimum range capability of the vehicle. This permits the pilot to home in on his landing site if it is within these maximum and minimum range limits by estimating the future trajectory from the existing angle of attack. In the case of stall and flare out, angle of attack provides feedback for their proper control.

Another important requirement, is for greater reading accuracy of gross altitude and velocity. With respect to the altitude and horizontal velocity displays which have been described, this would involve the expansion of the scale interval size on the displays of altitude below 100,000 ft and of velocity below 8,000 ft/sec.

A third requirement is a turn and bank indicator (lateral acceleration) to be used in conjunction with the attitude indicator. This will permit the pilot to make coordinated turns during adjustments prior to landing and at the lower speeds.

Not to be overlooked, of course, are the stringent requirements for flight path control during landing. What this means, in terms of display requirements for a re-entry vehicle configuration, has not been studied in detail. The attitude indicator which has been described has director needles and thus, is capable of being incorporated into an ILS system. Performance would be more adequate when a separate glide slope and localizer display was utilized. Another consideration, and perhaps the most adequate, is to provide for a GCA system and utilize this as the prime landing mode for a glide re-entry vehicle.

2.4.4 Integrated Crew Station Panels

To summarize the previous discussion of the control display subsystem, Figures II-2-19 to II-2-23 present the layout of the vehicle display and control panels so that the relationship of the functional areas and the displays and controls on them can be clearly seen. The relationship of these panels to one another can be seen in Figure II-2-4.

2.5 PROBLEM AREAS AND RECOMMENDATIONS

2.5.1 Areas Requiring Further Research and Study

There are 4 major areas where future research effort would be of principal benefit. These are: 1) providing experimental support to the designated steps of the design method, i.e., control-display parameter derivation, display specifications and display evaluation, 2) instrument mechanization and construction techniques, 3) optics, and 4) man-computer relationships.

2.5.1.1 PROVIDING EXPERIMENTAL SUPPORT

With respect to the first area, several areas of concern have been uncovered during this initial cycling period of the design method. These areas are associated with the display of navigational data during the midcourse guidance phase of the mission and the display of velocity correction parameters, also in the midcourse phase. It is believed that primary emphasis should be put on determining the adequacy of these critical parameters in terms of both information requirements and display variables.

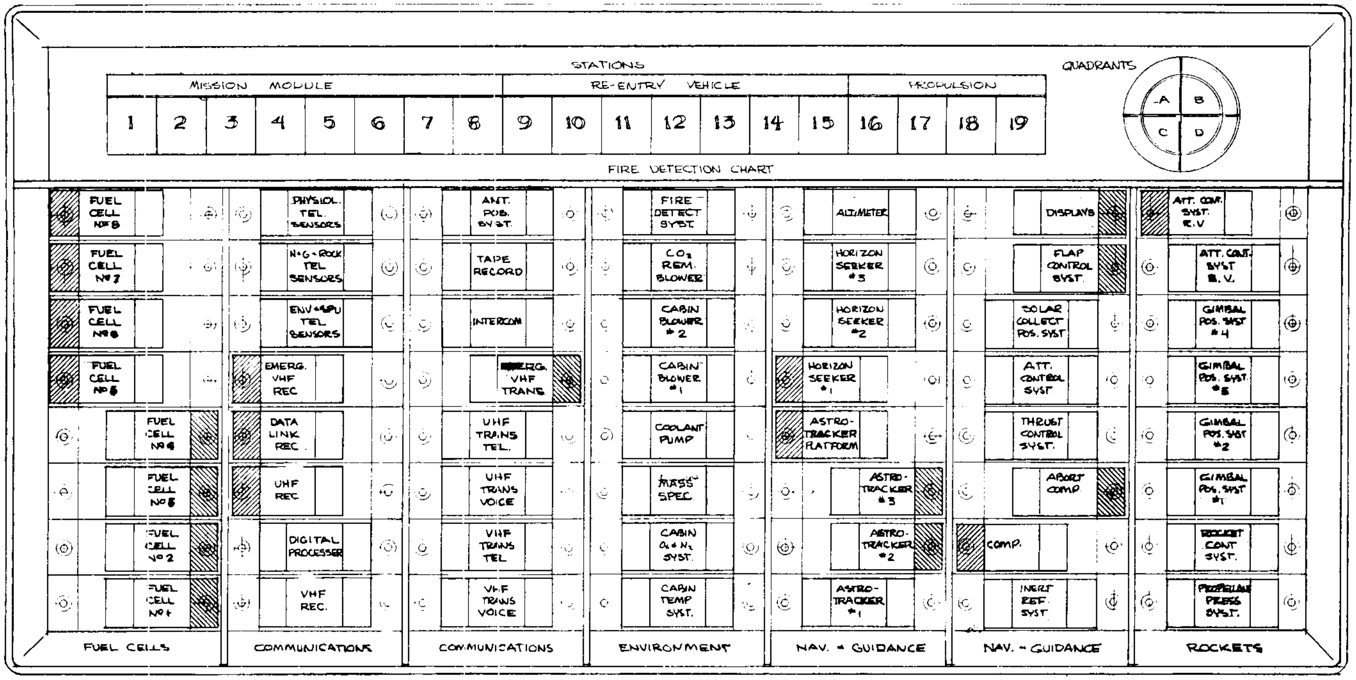

Figure II-2-19. Power distribution panel

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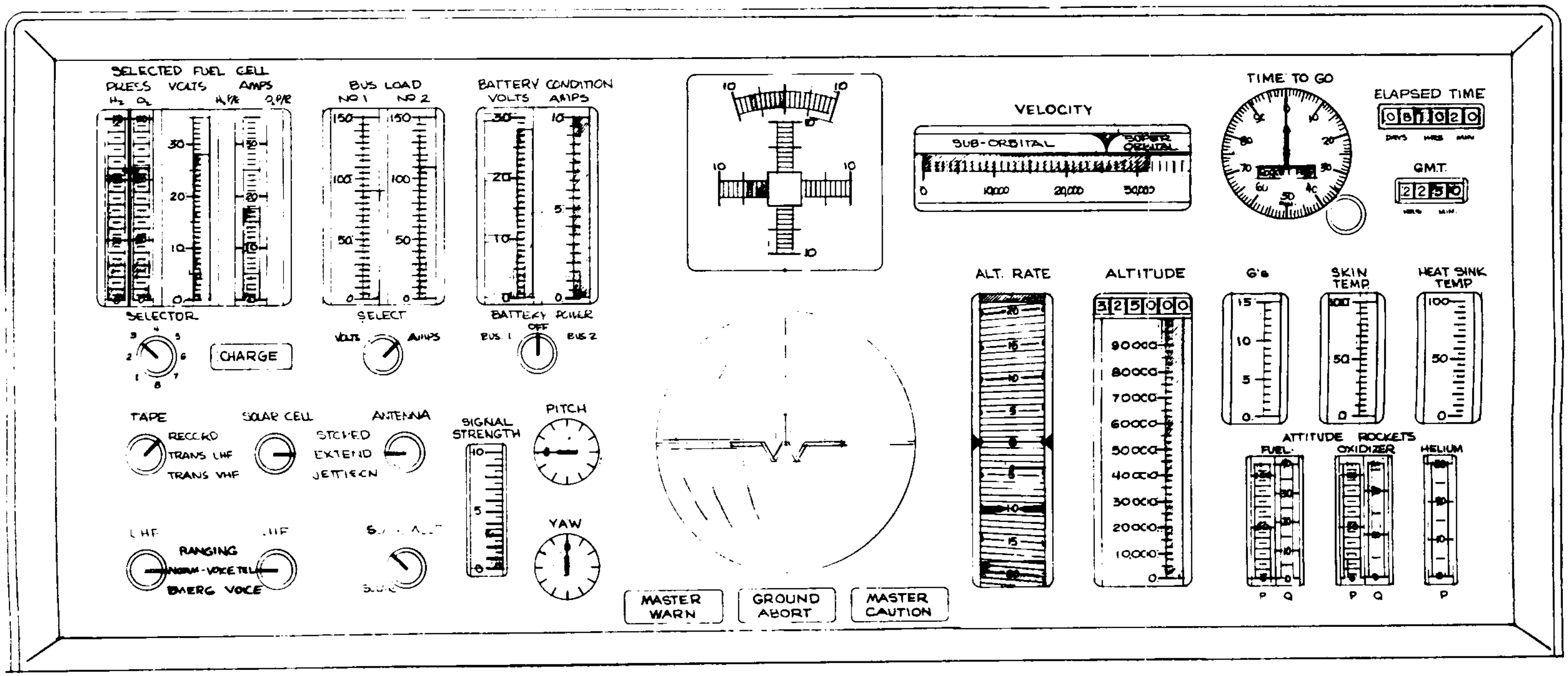

Figure II-2-20. Electrical power, communications and re-entry panel

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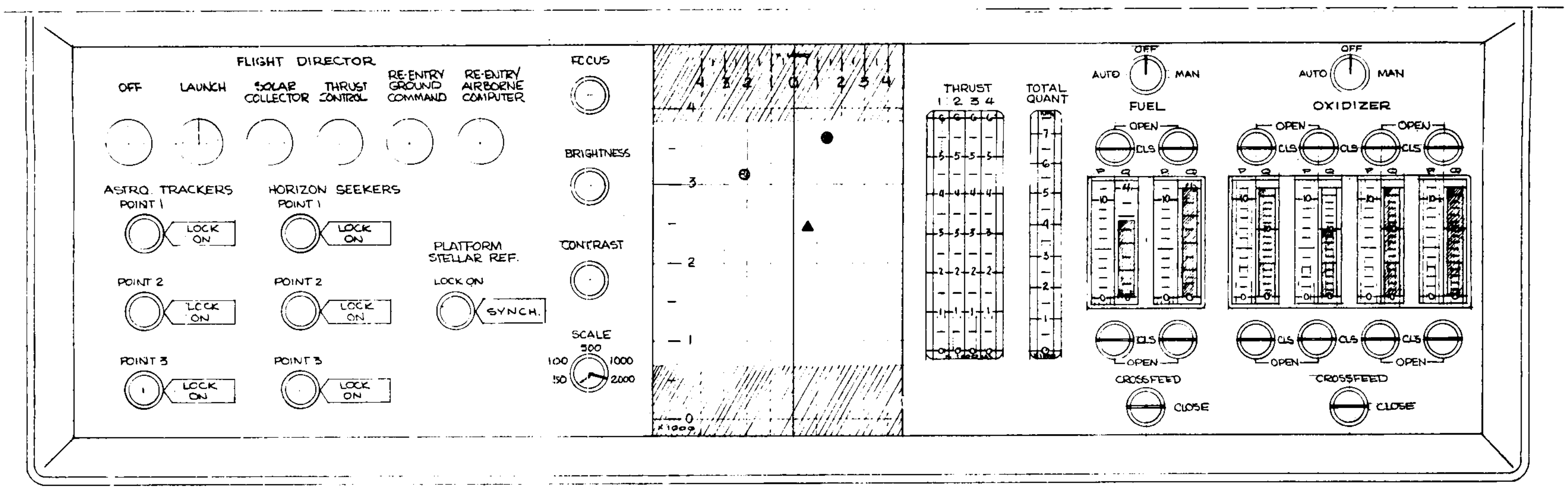
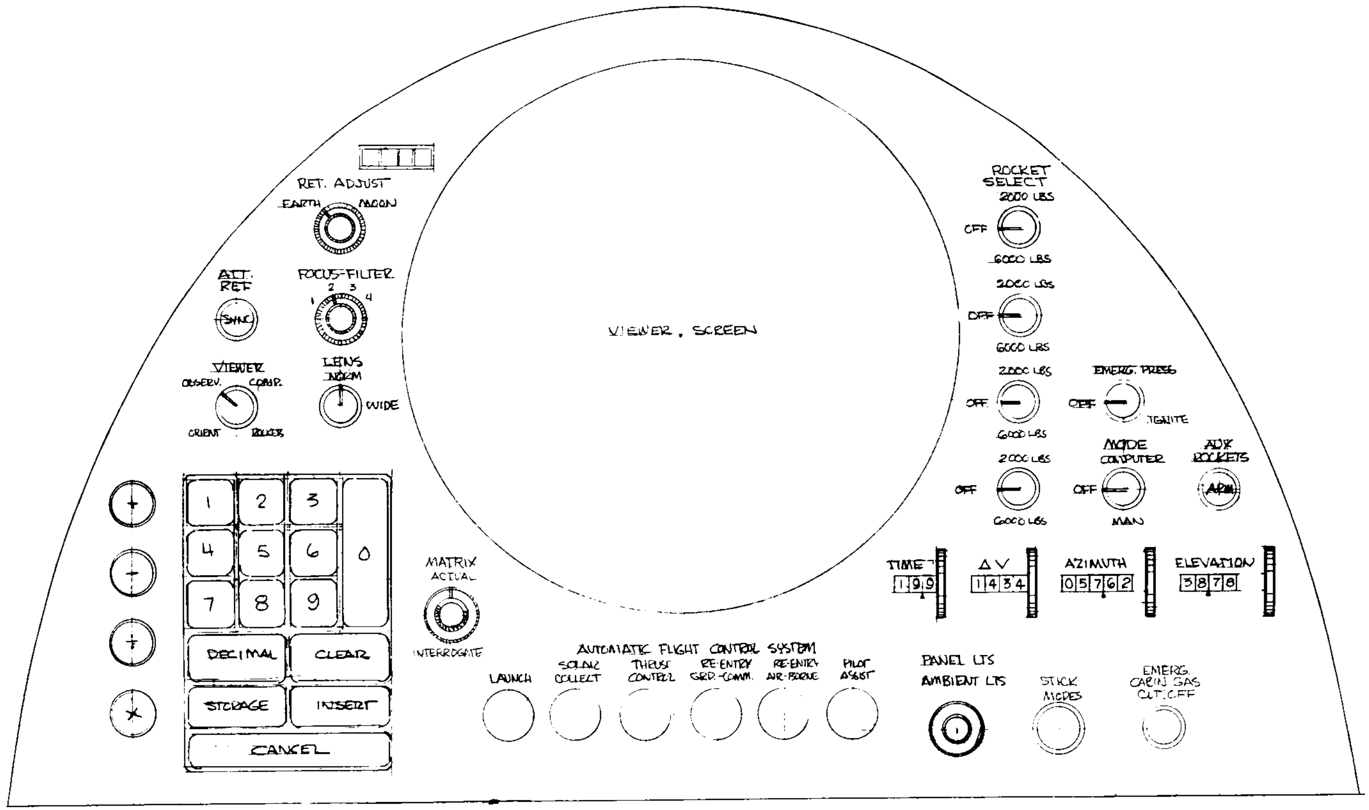


Figure II-2-21. Rocket power and flight director panel



OFF	EARTH AIM PT.
LAUNCH	RE-ENTRY
EARTH ORBIT	COMPUTE
SPACE COORD	SENSORS
LUNAR AIM PT.	DATA LINK
LUNAR ORBIT	ATTITUDE

OFF	AEROD CONT.
FLT. CONT. SYST.	ROCKET CHAMB
SOLAR COL.	ASTRO TRACKER
ANTENNA	HORIZON SCAN.

Figure II-2-22. Cocoon panel

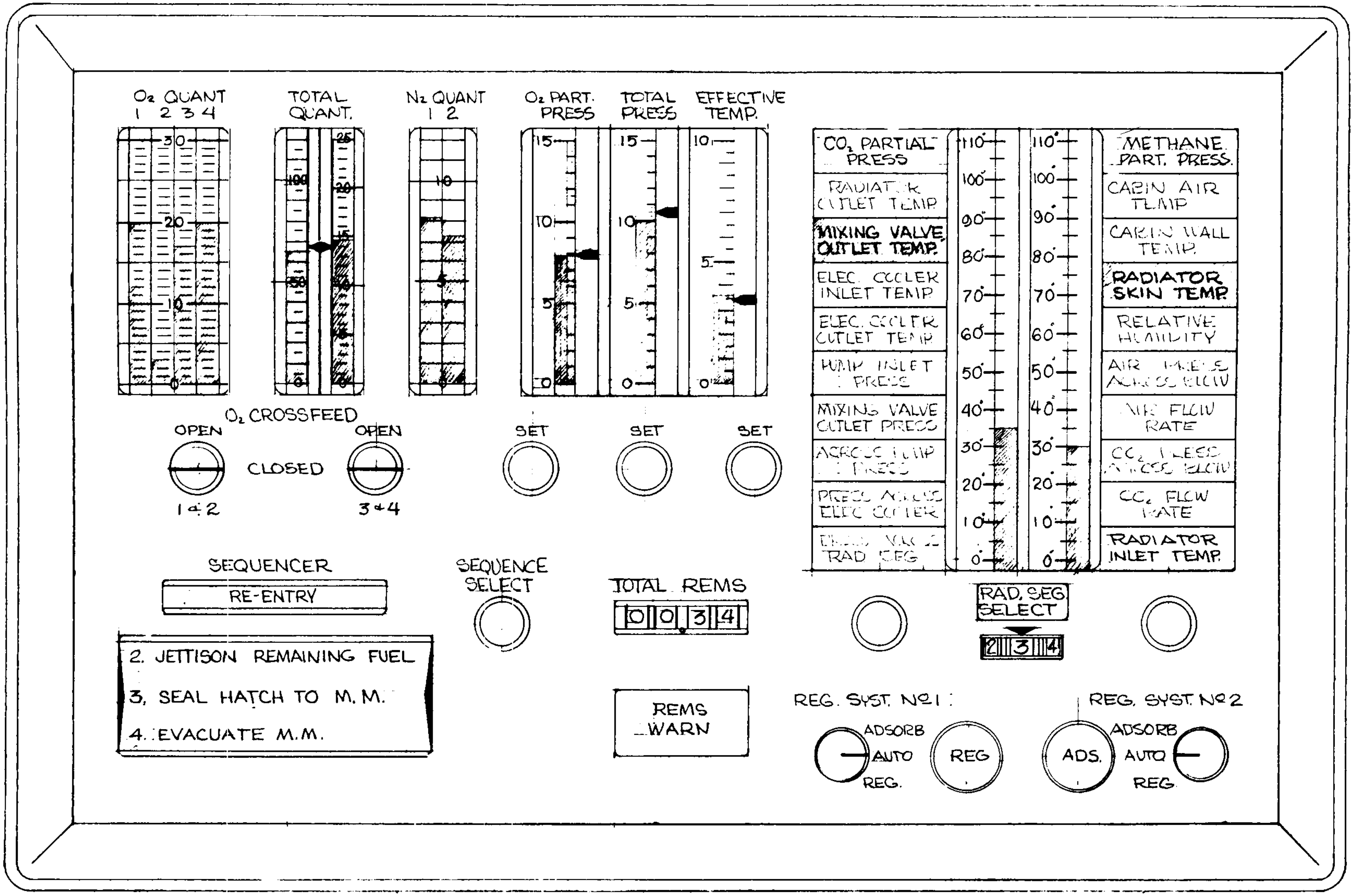


Figure II-2-23. Environmental control panel

Not to be overlooked, however, are the more minute system details which require much greater effort than has been expanded in this study program. These range from establishing the quickening function of moving elements within a particular display to broad system studies for determining system performance adequacy in the various modes of system operation. In summary, the basic consideration is that further recycling, especially detailing of the time line analysis and carrying out the experimental support and evaluation steps of the design method, is a requirement for the adequate development of the control-display subsystem.

2.5.1.2 INSTRUMENT CONSTRUCTION AND MECHANIZATION TECHNIQUES

Instrumentation techniques and state-of-the-art considerations will play an important role in the final Control-Display configuration. The approach taken here was to emphasize electroluminescent displays over other types of instrumentation because some of their characteristics are seen as inherently superior to other types of mechanization processes. For example, they have characteristics which may make them more reliable and durable than simple meter movement mechanisms. However, there are problem areas with respect to E-L displays and consequently study effort must be applied in developing the E-L displays which have been envisioned as being employed in the APOLLO system.

All electroluminescent instruments are of the solid state type and consist of replaceable modules for both display elements and electronic circuits.

1. Display Method - This method of display utilizes a combination of piezoelectric and electroluminescent materials. The resulting display is seen to be reliable and unaffected by environmental extremes to shock, vibration, and temperature. The moving display may be either a column or bar of light positioned in proportion to digital or analog inputs. Output light intensities are satisfactory for use under normal ambient lighting conditions.

2. Operation - The system envisioned utilizes a wedge-shaped stack of piezoelectric crystals excited by a variable frequency oscillator so that different crystals of the stack resonate at different applied frequencies. The oscillator frequency and output is controlled by sweep and gating circuits, respectively.

The gating circuit and its input determine the area or crystals to be excited. The electroluminescent phosphor is placed in intimate contact with one surface of the crystal stack and is excited by an increase in field strength generated by the crystals.

3. Advantages of E-L, P-E Display - This method of display offers a compact module with an apparent moving element which is obtained without the use of servo systems, complex mechanisms or heat generating lamps. Additional advantages are: low power consumption, high E-L lamp brightness, no moving parts, and low voltage requirements.

4. Areas to be Investigated - The piezoelectric crystals of the type required to produce this display operate at above 30 KC. E-L as observed in zinc sulphide phosphor increases in brightness with increasing frequency. The most efficient operation occuring at approximately 100 KC. However, investigation indicates a sharp decrease in brightness of this phosphor occurs above 200 KC. Also, presently available E-L phosphor ages rapidly at high frequencies. E-L phenomena have been observed in a number of different materials with various mechanisms of electroluminescence, but little is known about the high frequency characteristics of these materials. The fatigue properties of E-L materials other than ZnS are likewise unknown. However, it is likely that investigations of high frequencies with respect to these E-L materials will show that increased life is possible. Also circuits operating the E-L lamps can be gated and modulated to produce short duty cycles. This should improve life by a ratio inversely proportional to "on" time.

2.5.1.3 OPTICAL SYSTEMS

In the course of developing instrumentation for aircraft, there has been a shift from the dependence upon the vision of the man. The development of high speed aircraft and all weather operations have been attended by the replacement of man's visual sensing functions, especially by electronic means. This trend can be shown to be due largely to the intervention of extraneous objects between the observer and the objects which he wishes to see. A large percentage of the problems are brought about by the nature of the atmosphere and its changing conditions from time to time.

Optics and visual systems will become more attractive in space vehicles because there will be no atmosphere and no intervening objects to make visual systems unreliable. The designer will be free to exploit the fact that most of the desired objectives will be already radiating light energy and hence, will require no power for sensing. Since light waves are already emanating from the objects of concern the designers merely have to provide ways to reflect, refract, and filter the light to achieve his momentary objective. Light is generated without his providing any power and is transmitted already across vast distances in very accurate paths.

A periscope is an example of an optical system applicable to APOLLO. This device can be used, among other things, to determine altitude above the earth by measuring the angle which the earth occupies in the Mercury astronaut's field of view. It can also be used as an attitude sensor. Given the right kind of displays, the man will be able to determine quite accurately when he is pointing a reference line at the center of the earth. This optical attitude sensing can be used not only for close earth or Moon satellite operations but can be used for almost any coordinate system based on first magnitude celestial bodies.

More important to APOLLO than either attitude or altitude sensing is the potential use of optical systems in space navigation. There are numerous problems to be considered in the design of such equipment. Some of these have to do with the resolving power of the systems themselves. Others may be influenced by the fact that the APOLLO vehicle will be relatively small in mass and hence, may be easily perturbed. Other problems have to do with the possibility of automatic operation, normally with manual objective selection, and perhaps manual override. Position information thus obtained may be used directly for navigation or may be used to provide corrective inputs to an inertial navigating system. In any event the number of variables involved and the importance of the area recommend a study of optical systems as being of primary concern for the APOLLO development phase.

2.5.1.3.1 <u>Man-Computer Interface</u>

The design objectives for APOLLO call for the human operator to be in command of the mission. This implies that he be provided with those items of equipment which will give him support for routine functions. The preliminary study of APOLLO design shows that a large capacity onboard computer will be required. In order for the human operator to be in command, this computer must be capable of operating in several different modes. Its input and output equipment must be arranged so that the operator may elect his decision strategy and select the level at which he wishes to use the computer in the control function.

In the preliminary study most of the computer modes assume that the computer receives continuous input from the inertial integrating system. With these inputs it keeps a running account of trajectory influences to show present position and orbital elements. It may be interrogated by the operator to show future positions and how to get to desired orbital positions. Upon request it will read the results out in either Moon-centered or earth-centered co-ordinates or perhaps a co-ordinate set around the Moon initial aiming point. It can be made to show required corrections in terms of the velocity increments with pitch and yaw angle specification, and further transform the velocity increment to boost control terms. It may be entered with a proposed boost program to predict before firing the effects of the proposed boost on the resulting trajectory. In a quite different mode the computer will take angular sightings of pairs of celestial objects plus precisc time and calculate space position for midcourse correction of the inertial system. It is also envisioned that the operator may want to have input and output equipment to use the computer for general purpose computation. This would imply a capability of adding, subtracting, multiplying, taking square roots, etc. In the final mode this same computer will act as a re-entry energy management computer which accepts inertial, atmospheric, and stored inputs to calculate a probable landing site and the controllable variable range about that site.

Input equipment to the computer required for this kind of operation include a mode select panel, an address panel, a digital keyboard panel, and inputs from the inertial sensing system and from atmospheric sensors on re-entry. Outputs of the computer to the human operator would be visual, primarily by means of a charactron-type cathode ray tube or suitable solid state devices. It is expected that a part of the displayed parameter outputs of the computer are visible at all times. These will include those characteristics of the orbital elements which are to be used in programming controllable boost. Additional display space will accommodate whatever predicted and interrogated values may be requested of the computer. The input panel to the computer will allow the operator to identify the conditions for problem solution and direct the specific question it wants answered by the computer. Two kinds of predicted information are required. One kind would be that having to do with some future state if no further boost is required. The other kind would be a fast time calculation of the consequences, at some particular time, of boost at some particular time.

Actual orbit, desired orbit, and predicted orbit may be computed either for earth-centered, Moon-centered, or aiming-point-centered co-ordinates as determined by mode select. Further study may show that it is desirable to have these calculations appear in two or more co-ordinates of the same time. This will be determined by the particular pattern of usage which the computer will have in the operator's control over the vehicle, which is itself a subject of a part of the development program envisioned.

It appears clear that a large capacity computer is required for APOLLO success. Furthermore, it appears that the control which the human operator exercises over this computer will be a large part of the determination of the flexibility of the operator's role in the APOLLO mission. Since the computer must have enough capacity to calculate very complicated trajectories it seems that very little more would need to be added in order to allow the operator to have any kind of computing capability which seemed useful for him in controlling the mission. However, just what things are needed and just how the input and output equipment should be designed in order to serve these needs are questions which will require considerable attention in the development phase.

3.0 Maintenance and Reliability

3.1 INTRODUCTION

Maintenance, in the most general sense, involves a set of operations, performed automatically or manually, which keep a system at a desired operating level. Although intimately involved with considerations of reliability, maintenance is different primarily because it does not directly involve the design of operating components, but instead concerns a set of procedures employed when adequate reliability is not forthcoming from the hardware.

The problem of design for maintenance depends to a large degree on a thorough knowledge of component, subsystem, and system reliability. Where design limitations result in a failure to obtain required levels of reliability, maintenance procedures may be employed to achieve requirements.

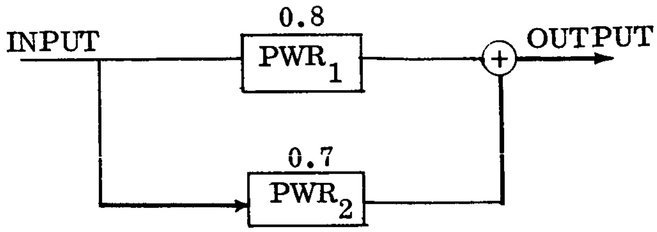
In addition to the reliability analysis, maintenance design demands further study of the systems involved, the nature of the mission, and the particular constraints which exist for various phases of the mission. With this information available, specific maintenance design criteria and operational procedures can be evaluated and applied in terms of overall system requirements.

The major portion of the following discussion will concern itself with in-flight maintenance. However, the requirement for maintenance relates to all phases of system development and preparation.

3.2 RELIABILITY, MAINTENANCE, AND SYSTEM DESCRIPTION

The purpose of this section is to illustrate the relationships among the three concepts in the title: reliability, maintenance, and system description.

It is commonly accepted that system reliability, or the probability of in-tolerance operation, is a product of component reliabilities. An important exception to this statement is redundant hardware. For example, the probability of power output is a function of primary and secondary power supply reliabilities, where, schematically, is used to turn on the secondary supply if the primary supply fails. Similarly , if we take to imply readout and switching this combined function can be referred to as a shunting unit. Then



describes the use of a shunting unit; and the probability of the output being in-tolerance is expressed by:

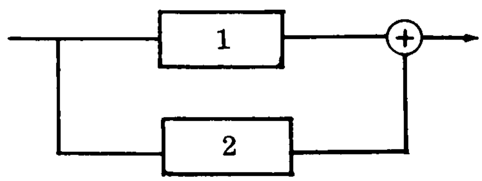
$$(0.8) + (1-0.8)\ (0.7)$$

if the probability of the input is 1.0 percent and the probability of shunting is 1.0 percent.

If the probability of input is 0.6 percent and the probability of shunting is 0.9 percent, then the probability of the output being intolerance is described by:

$$(0.6)\ (0.8) + (0.9)\ (1-0.8)\ (0.7)$$

It is important here to note two things: (1) a secondary or backup hardware unit adds to the reliability of a system, and (2) a readout is useless unless it acts as a shunting unit (or supplies the information indicating that shunting is necessary) to add in the redundant mode. Therefore, a good design requires that a readout not be inserted unless it results in an action contributing to reliability. Conversely, an additive unit or action must be provided if is provided. This design mandate applies to maintenance actions as well as to redundant hardware since maintenance is always additive, i.e., adds to reliability. To illustrate this, let us take paralleled hardware in a system.



If the reliability of Box 1 is 0.8 and the reliability of Box 2 is 0.6 and the reliability of shunting* is 0.9 then the probability of output is (0.8 + (0.9) (1-0.8) (0.6) = 0.908. Now suppose that Box 2 describes a corrective maintenance action instead of a parallel hardware unit and suppose that the 0.6 percent is the probability that corrective maintenance of Box 1 will be accomplished before total system failure occurs. It is extremely important to note that the description above is appropriate for describing maintenance performance as well as hardware performance and shows that maintenance performance is additive. Again, it has been shown that a $-\oplus\rightarrow$ is needed to turn on corrective maintenance just as it is needed to turn on parallel hardware.

A requirement for designing a system of high reliability should now be apparent: human performance which affects system operation must be accounted for in a system description. This is equally true for critical human performances which are not additive maintenance performances; critical in the sense that the human operator is used as a component where system reliability is a product of component reliabilities. Such multiplicative human performances can and must be accounted for in a system description.

The foregoing discussion has suggested that in designing for high reliability:

1. The system should be described in terms which show all operator and hardware performances required to get the system to operate. These are the critical or multiplicative performance.
2. Estimates must be made of reliability of each multiplicative performance.
3. If any multiplicative performance reliability, either hardware or human, is too low to permit the realization of the desired overall system reliability, the designer must attempt to improve the unit performance (a) for the hardware by selecting

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* This example neglects to partition the performance of the shunt into detection of adequate performance of Unit 1 and detection of inadequate performance. These may have different reliabilities. A more precise treatment would have to allow for the four contingencies of possible performance of Unit 1 and decision by the shunt. These calculations are also simplified by the assumption of independence of the component reliabilities.

"better" units, derating, etc., (b) for operator performance by better training, use of job aids, human engineering of the interface, or reallocation of function to hardware. (It may be noted that several of these techniques are used as a matter of course, even with subsystems normally having high reliability.)

4. Then when the reliability of each multiplicative component has been maximized in time, cost, weight, etc., obtain the needed additional reliability by providing for additive performance, either by switching in redundant hardware and/or by performing maintenance operations.

The previous discussion has emphasized the utility of employing man to perform maintenance function and has essentially suggested his use when system reliability is not available from hardware. It is important to realize, however, that the same type of analysis which permits the understanding of reliability aspects of the system also serves to indicate where man can be used in place of hardware because of his special capabilities.

3.3 DEVELOPMENT OF IN-FLIGHT MAINTENANCE STRATEGY

3.3.1 General Considerations

The aim of in-flight maintenance is to increase the reliability of the APOLLO vehicle by making use of the capabilities of the crew. In doing this, the crew faces some problems that are common to all maintenance operations in addition to problems that are unique to maintenance in a closed system under severe constraints.

Maintenance functions are ordinarily divided into preventive and corrective. Corrective maintenance, when undertaken without the special constraints of a closed system vehicle, includes the following:

1. Detection of inadequate performance
2. Location of the source of inadequate performance
3. Determination of the action needed for repair

4. Possible provision of substitute performance during the interval required for repair
5. Execution of the repair
6. Return to normal operations

3.3.2 Considerations Peculiar To The APOLLO Vehicle

APOLLO, with severe constraints and extremely stringent requirements for the maintenance of its functioning, accentuates certain requirements for maintenance performance which stem from the fact that the system is closed and its resources are limited. The accomplishment of maintenance functions is no longer simply a matter of determining what is wrong and what is needed to effect repair by a maintenance crew. Maintenance must be analyzed and synthesized in the context of other requirements of flight and the maintenance tasks must be allocated among other activities and task requirements.

It is clear that the effects of a malfunction, the seriousness of ignoring it, and the difficulty of effecting a repair, are all determined by the context within which they occur. For instance, the activities in which crew members can engage will be radically altered depending upon whether the time considered is during an interval of weightlessness or one of boost acceleration; e.g., considerable peaks of activity may be anticipated just before, during, and just after midcourse correction but comparatively lighter work loads are anticipated before and after this period.

Simultaneous failure of components in ground maintenance can usually be handled by the addition of another maintenance crew. Previous failure records of malfunctioning components can lead to a change in spare parts inventory. But additional men and a multiplicity of parts are not consistent with a limited size closed system. Furthermore, maintenance is affected by circumstances affecting the crew but not the hardware — these contingencies may be termed "crew disablement" and may be a function of such things as health, requirement for wearing a pressure suit, or the effect of G-forces.

Phase of flight affects the actions that can be taken, and hence the long-run consequences of a malfunction. A defect in the gaseous environment control subsystem in an early stage of the flight should receive considerably higher priority than the same malfunction occurring just prior to the re-entry stage. The same malfunction occurring during the initial acceleration phase could be treated only by whatever action has been programmed automatically, or by an action which could be initiated by the crew while they were relatively immobilized by the restraint system. A serious malfunction at this time might have disastrous consequences whereas the same malfunction occurring later could be treated with ease, perhaps requiring a change in the date of the return flight but no fundamental alteration of the mission.

Within the closed APOLLO system, maintenance functions must then include these additional steps:

7. First-approximation assignment of priority to corrective action (in the light of the consequences expected for the mission, the phase of flight, past malfunction history, the load of other duties at the time, etc.)
8. The specific nature of the malfunction consequences must be determined.
9. Remedial steps (or none) must be considered and selected by comparison of the relative effects upon the mission of carrying out that repair or engaging in other duties. A statement of what appear to be the alternate classes of remedial action is as follows:
   a. Repair: return component to acceptable operating condition by restoring original parts to "as new" condition (e.g., tighten, seal)
   b. Replace: provide a substitute part or mode of operation through
      1) module or component replacement from "stock"
      2) switching to equivalent substitute
      3) switching to functionally equivalent alternative mode
      4) "pirating" of substitute part from lower priority subsystem. This procedure will require priority tables and substitution charts for all critical components during in-flight mission phases.

c. <u>Delete</u>: eliminate an operation or function to prevent impending malfunction

d. <u>Degrade</u>: operate component at less than rated level, and/or intermittently.

e. <u>Prevent</u>: detect incipient malfunction via calibration or operational checks and prevent by one of above modes as appropriate.

Thus, the detection of malfunctions must take into account, to an extent greater than is ordinarily done, the context in which malfunction occurs. Decision making, calling for the selection of alternative modes for effecting adjustment, or even of ignoring the malfunction become of central importance. There will be certain emergency situations so serious and requiring action so prompt that the design of equipment and the training for its use should permit immediate response to them. These extreme emergency situations should be sharply differentiated from those allowing more considered allocation of resources.

The following paragraphs contain discussion of some developmental steps which might be taken to arrive at an integrated strategy for the decision-making processes necessary to APOLLO in-flight maintenance.

First delineate the mission profile in terms of its various phases. For each phase, the next requirement will be one of reviewing, for each subsystem in turn, the malfunctions which might develop for that subsystem.

On the basis of this list of malfunctions classified by mission profile phase and by the subsystem involved, the next step becomes one of determining the extent to which a malfunction in a given subsystem will preclude or degrade the successful operation of that subsystem. This is followed by a determination of the criticality of that subsystem for the particular mission phase under consideration. Top priority to corrective actions thus can be assigned in terms of malfunctions which occur to the most needed subsystems during each phase of the mission.

In this connection, serious consideration should be given to the concept of a maintenance integration program which combines the idea of priority with that of sympathetic or modular unit design. This concept was originally developed by General Electric as a technique for automatically switching functionally identical sub-assemblies from a low priority system to a critical system. The concept lends itself very well to manual implementation by the crew, aided by priority schedules in list or book form.

As an additional step in considering the consequences of malfunction and priority assignment, it will be necessary to deal with certain interactions which occur as a result of particular malfunctions. For example, a malfunction in the oxygen supply system may be of relatively little importance to mission success at the moment of occurrence; however, if it is allowed to exist for a period of time, the crew may develop hypoxia. This in turn could degrade performance for subsequent corrective or any actions to a dangerous point. Malfunction consequences of this type must be carefully accounted for.

The next step involves human factors personnel working in close conjunction with engineering design personnel to conduct trade-off analysis. These trade-off analysis must consider carefully each malfunction and list possible remedial steps which are broken down into those having equipment design implications and those having crew training implications. The appropriate philosophy will be one of initially assigning all corrective in-flight maintenance to the crew. But there will be instances in which crew performance capability will either be not possible or will be seriously degraded. In these instances, equipment design must account for the maintenance capability.

In the course of development, each item of hardware making up the subsystems and systems of the APOLLO vehicle will have to be considered from the point of view of the maintenance functions such as localization, isolation, removal, calibration, repair, and/or replacement, etc.

As stated above, a separate analysis should be performed for pre-flight maintenance and for in-flight maintenance.

Items of information which must be collected include:

1. All operating parameters which <u>can</u> be tested and all operating parameters which <u>must</u> be tested.

2. The range, critical value, tolerance, accuracy, or stability requirements for each parameter.

3. The dimensions of measurable aspects of each parameter.

4. Test equipment requirements. Particular attention should be paid to this with respect to in-flight test equipment required.

5. Other equipment, such as handling equipment, transportation equipment, and holding fixtures required for maintenance.

6. Special facility characteristics required for maintenance, e.g.,

   a. work space
   b. ambient air conditions
   c. vibration control
   d. illumination
   e. power
   f. noise control
   g. structure

7. The effect of tests and malfunctions on each item, including such charactistics as tolerance of electrical, mechanical, and thermal stress; the effect of test or maintenance on time item will operate accurately; effects on item stability.

8. The effect of testing or malfunctioning on associated items which can be affected, e.g., overstressing or dependent failure.

9. Number of man hours required to test and/or maintain the item.

Table II-3-I presents in summary form an initial analysis of the factors mentioned in the preceding paragraphs. For all phases of the mission except prelaunch, the emphasis has been placed on maintenance activities which can be performed by the crew.

TABLE II-3-1. MAINTENANCE MODES AS A FUNCTION OF MISSION PHASE

	Pre-launch	Launch	Midcourse (Outbound & Inbound)	Lunar and Midcourse Corrections	Lunar Orbit	Re-entry	Recovery
AIM: Major Objective of maintenance action during particular mission phase.	Predict Inflight failures. Detect existing failures. Correct to permit mission initiation.	Detect Malfunction for emergency corrective procedure.	1. Detect Malfunction 2. Perform corrective action to permit mission continuation. 3. Prevent impending malfunction. 4. Check out subsystems to be used later in mission.	1. Detect Malfunction 2. Perform corrective action to permit course correction to occur.	1. Detect malfunction 2. Perform corrective action to permit mission continuation and transfer to earth.	1. Detect malfunction 2. Perform emergency corrective actions to complete re-entry.	1. Detect malfunction. 2. Perform corrective action to permit survival, detection, recovery.
ACTION: Alternate techniques for implementation of maintenance aim.	Repair Replace	Switch in Substitute (manual, auto) Abort Secondary mission	Repair Replace (Module, "Pirate", Switch) Delete Degrade Prevent	If critical to ΔV use manual or automatic substitution. If non-critical employ techniques as indicated for Midcourse.	SAME AS MIDCOURSE	Manual or automatic substitution	Repair Replace Prevent Degrade Delete
MODES: Aids and procedures to assist maintenance.	1. All standard diagnostic equipment. 2. Computer based checkout via umbilical. 3. Simulated inputs to onboard system.	Rapid manual or automatic override substitution. Diagnostic & warning disp.	Tools (Special Design) Substitution and priority scheme. Troubleshooting/diagnostic job aids. Calibration & Oper. check procedures. Diagnostic & Warning display. Replacement modules. Manual/auto - override/substit. Special training.	Rapid manual or automatic override/substitution. Diagnostic/warning display.	SAME AS MIDCOURSE	Rapid manual or automatic override/substitution/Diagnostic Warning display	Tools, job aids Substitution & priority schemes. Replacement. Diagnostic Warning Displays.
CONSTRAINTS: temporal environmental logistic	Launch Time Requirements (allowable hold)	High "G"/Temporal restrictions All action from "restrained" position.	Weightlessness Somewhat reduced temporal restraints. Accessibility Number of Spares	Time Restrictions "restrained" positions	SAME AS MIDCOURSE	High "G" (especially after weightlessness) Heavy crew loading. Tight time restriction.	Extreme climatic/environmental extremes (temp./terrain)

As indicated in the table, within each of these phases, it is possible to define certain maintenance requirements that are imposed as a function of the characteristics of the phase. It is also possible to make a determination of the subsystems that can be expected to be operating during the phase and thus to define within limits the type of maintenance activity that may be required.

During the pre-launch phase, major concern will be with final assembly of the vehicle, performing a complete system check to predict any potential malfunctions and to detect any existing malfunctions and, finally, to take corrective action. Particular concern should probably be oriented toward potential malfunctions that may occur during the boost phase. During this phase of activity, maximum utilization of automatic checkout equipment and of simulated inputs to all of the subsystems can be made. Maintenance activity would include replacement of all marginal and non-operating components in the entire system.

During the boost phase, maintenance activities would be restricted to the detection of malfunctions and marginal operating conditions through monitoring by ground equipment and by crew members. For this phase and for the re-entry phase, malfunctions indicators and to the extent consistent with safety and weight considerations, switches for corrective actions should be mounted on a panel that is accessible to the restrained crew during high G maneuvers. It appears at present that these may be limited to the propulsion, life support and communications subsystems, and onboard abort command or back-up mission. Because of environmental stress, it would be virtually impossible for the crew to perform any active maintenance work. Most malfunctions occurring during this phase would have to be met with a decision to abort the mission or to continue with a degraded mission capability depending upon the severity of the malfunction and its potential effect upon the survival of the crew.

During the lunar-bound, lunar-orbital and earth-bound free flight phases, continuing system checks would be performed by the crew. During the outbound trajectories and lunar orbit, primary consideration will be given to the detection of malfunctions, appropriate remedial action, or a decision to continue with a degraded mission capability. During the inbound trajectories, increased emphasis will be given to prediction of potential malfunctions that would occur during the re-entry phase or the recovery phase. This becomes of particular importance since during the re-entry phase the high environmental stress will prohibit any active maintenance activity and serious malfunctions, particularly in the guidance/control system, may result in destruction of the vehicle and loss of the crew. Finally, during the recovery phase, the maintenance related activity would be to insure the integrity of the survival subsystem.

The final vehicle design will be one which provides the crew with the capability of correcting many of the potential malfunctions. The remaining malfunctions and accompanying corrective actions will be those which the analysis has indicated would be inappropriate for assignment to the crew. For those corrective actions which are assigned to the crew, careful account must be taken of the requirement for immediate and extremely accurate performance. Where such requirements are found to exist, a requirement for good human engineering design and extensive training is generated. The conditions under which this training is provided must be such as to account for the operation of any environmental forces which might tend to disrupt these performances during the period of actual flight.

3.3.3 Developing a Maintenance Strategy

One reason for including humans within space vehicles can be found in their decision-making abilities. They can sense and assimilate many kinds of system-status cues and can use this information in arriving at decisions regarding proper courses of action. Although this capability within the vehicle is of extreme importance, it is not wise to overburden it. To the extent possible pre-programmed maintenance strategies should be developed. Thus, for any given malfunction, knowing the mission profile phase and other relevant information concerning the time of its occurrence, a carefully developed plan of corrective actions should be available. The preparation of such strategies will be of unquestioned benefit to vehicle operators and also can be used profitably in assisting engineering personnel in equipment design reviews.

Means must be chosen for insuring the selection and proper utilization of the strategy. A beautifully designed maintenance strategy will be of little value if it is not performed as designed. Prepared in conjunction with the strategy must be means to insure that the strategy will be used appropriately. Physical guides such as check lists or job aids represent one means of allowing the strategy to be followed. Some strategies may be learned by the crew to the point where performance is virtually automatic in response to the appropriate set of conditions. In addition, since maintenance activities occur within a dynamic context, the influence of the maintenance strategy upon the mission program must be assessed and incorporated into the strategy.

Competition of normal mission activities with those required by the maintenance strategy must be assessed. A maintenance strategy represents a program of activities required of the operator. Care must be taken in order that the accomplishment of the maintenance strategy will not interfere with other activities required as a normal part of the mission. It would be on the basis of such competing demands that certain maintenance strategies would involve temporary delays prior to the activation of the strategy activities.

It should be pointed out that adequate human engineering of the operational equipment and test equipment utilized by the crew will contribute substantially to a reduction of time and errors and increased reliability of maintenance performance. The application of human engineering in equipment design is a substantial area in itself and will not be discussed further.

Job aids may take many forms but we are here concerned with procedural instructions for the crew which might be prepared in the form of technical manuals. The preparation of technical manuals is a unique specialty, requiring the attitude of the human factors specialist who is concerned with personnel performance, rather than engineers who are concerned with hardware performance. In preparing technical manuals, the specialists must exercise ingenuity to obtain complex job performance utilizing only common performance capabilities of human beings, and materials which provide for integration of these capabilities into specific required job behavior. In the operational situation, it is not the manual per se which is required but rather it is the crew performance which is induced or enhanced by the manual. The main requirement for manuals containing procedures or instructions is that they be prepared with a specific job in mind rather than specific equipment in mind.

Another function which manuals must serve is to provide information to the crew in a form which may be readily understood and utilized. Adequate data are available with respect to presentation principles of printed materials. These data are concerned with such things as legibility, readability, style of type, type form, type size, line length, contrast, and spacing. The effect of these variables is undoubtedly of some importance and should not be minimized. They may not, however, be the critical variables in preparation of procedures or instructions. More important is the logical design of the maintenance aids which permit man to perform a continuous sequence of contingent actions which are the result of decision-making processes.

To be concerned with crew members as decision-makers requires concern with the content, form, and adequacy of the information on which crew members make these decisions. Relating decision making to the problem of procedures and instructions, it is apparent that the crew is asked to perform certain actions, evaluate the consequences, and take further action on the basis of these evaluations. Procedures and instructions must provide clear descriptions of the actions which must be taken, a complete specification of the alternative results of these actions, and a course of action based on the alternative which allows of a minimum of uncertainty.

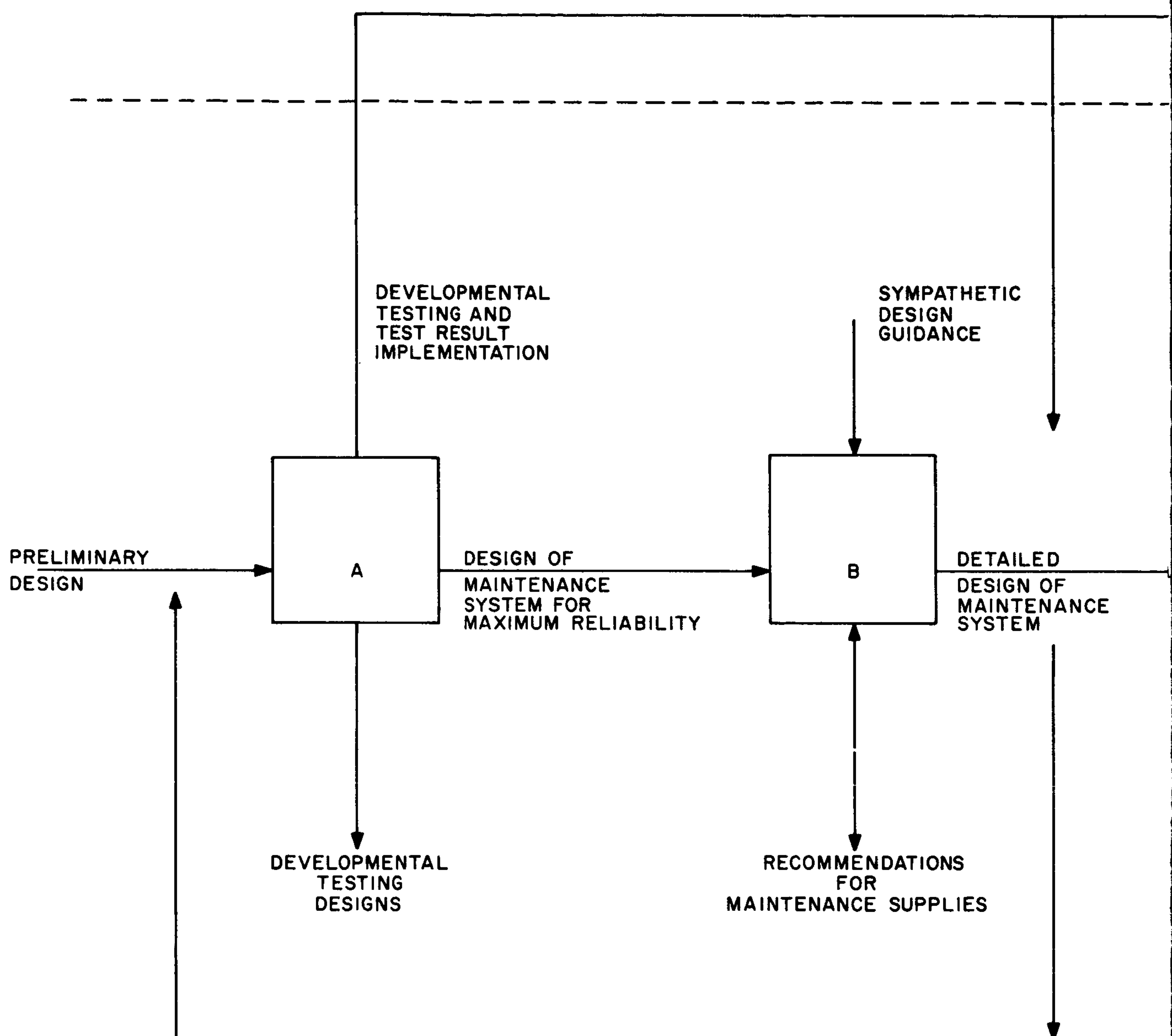
Suppose now that the problem is to determine what readouts are needed for maintenance of a subsystem. If the overall reliability of the system as it stands without any readouts equals or exceeds the system reliability that is required, then there is no requirement for readouts because readouts are used only for the purpose of introducing maintenance action which adds to overall system reliability.

Assume, however, for the purpose of example, that the overall reliability of the system without readouts and without maintenance actions is less than the reliability that is desired. The first step that should be taken should be to determine approximately the reliability of each of the functions within the system. When this has been done, each of the multiplicative units should then be examined to determine which of the units contributed the most to overall system reliability degradation. Among the multiplicative units, these will be the units with the lowest reliability. Effort should then be made to increase the reliability of the weakest links by engineering techniques. This can be done by attempting to improve the inherent reliability of a functioning unit or by adding a secondary or back-up unit with an automated shunting unit in order to add in the required amount of reliability for the paralleled functional hardware unit. Assuming that even after this has been done that the required overall system reliability has not been acheived, or, alternatively, that the weight penalty which results if an engineering approach is used to obtain the improvement of overall reliability is prohibitive. Then it is advisable to attempt to solve the problem by designing maintenance actions into the system to add the required amount of reliability. Clearly, in order to do this, the functional units which account for the greatest amount of overall system degradation must be identified because it is to these units that additional reliability through programmed maintenance actions must be added. At this point in the development process, there are two kinds of programmed maintenance actions which should be kept in mind: preventive maintenance and corrective maintenance. Of the two, preventive maintenance is generally more desirable first because repair is possible at non-critical periods, and second because by its very nature it usually does not involve down-time of functioning equipment. The down-time involved in corrective maintenance sometimes makes this type of maintenance impossible as a method of improving overall system reliability even though it is otherwise feasible. Scheduled servicing or scheduled replacement must be considered as a preventive maintenance technique to add in the required amount of reliability for the weakest functional units. If the overall required system reliability cannot be achieved by means of typical hardware engineering solutions or by means of preventive maintenance techniques, then corrective maintenance must be employed.

Before considering the selection of specific readouts for troubleshooting a subsystem, it should be noted that there is a requirement for a readout of the toal subsystem operation which can be used as a signal by the crew to turn on back-up systems, to go to another mode of operation, or to head for home.

Given a signal or set of signals which can be used to detect whether or not the subsystem as a whole is operating in tolerance, and assuming that corrective maintenance to achieve the desired overall subsystem reliability must be employed; what readouts must be provided so that corrective maintenance action can be taken when the overall system readouts indicate a failure? Again those multiplicative units with the lowest unit reliabilities must be focused upon and an attempt made to add enough reliability to each of these to bring up the total reliability to the desired level. This means that readouts must be provided which will enable the crew quickly to identify failures of the most likely to fail multiplicative units. (Note that this is different from saying that readouts must be provided which will enable the identification of failures among any of the most likely to fail units. Frequently some of these units will be additive units and identification of such failures for the purpose of performing maintenance may have much less effect on overall system reliability than the provisioning of maintenance actions for less likely to fail multiplicative units.)

Figure II-3-I presents in summary form the basic maintenance design program presented in this discussion, and indicates clearly the interactions which exist and the products the program should produce.


DEVELOPMENTAL
TESTING AND
TEST RESULT
IMPLEMENTATION
SYMPATHETIC
DESIGN
GUIDANCE
PRELIMINARY
DESIGN
A
DESIGN OF
MAINTENANCE
SYSTEM FOR
MAXIMUM RELIABILITY
B
DETAILED
DESIGN OF
MAINTENANCE
SYSTEM
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FOR
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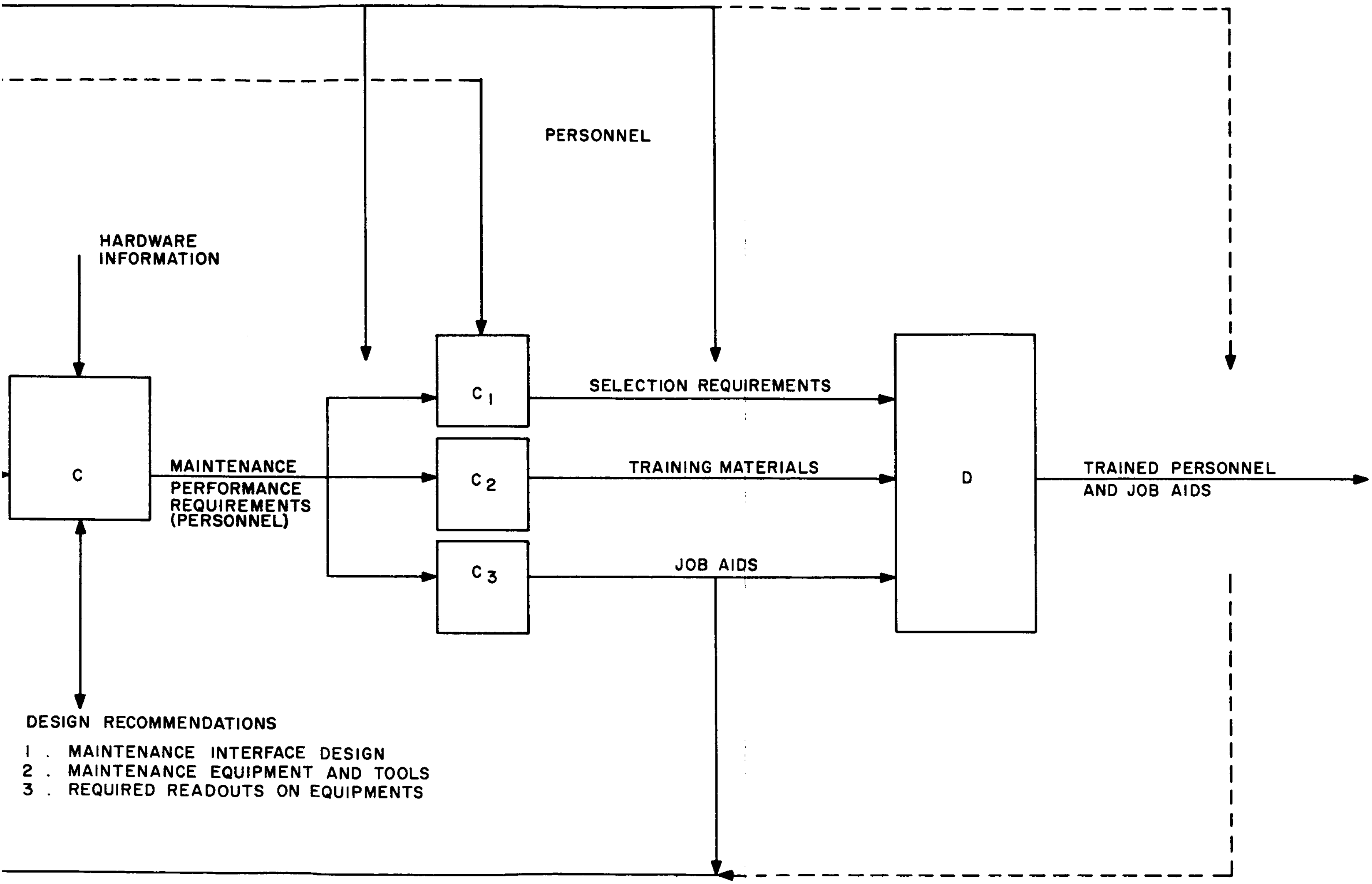

Figure II-3-1. Maintenance design program

4.0 Crew Selection

4.1 GENERAL CONSIDERATIONS

The most striking difference between the Mercury and the APOLLO program is the breadth of the APOLLO mission. In addition to the Mercury functions of piloting, observation and reporting, the APOLLO crew will conduct experiments during the orbital phase, play a role in vehicle maintenance and will be critical to the scientific success of the mission.

It is most probable that prospective crew men will be volunteers. Since the crews must be highly organized, maximally trained and disciplined, they will probably come from military organizations. Because of pilotry requirements, they will probably come from the various air forces.

The initial selection of candidates should be regarded as a screening procedure. The subsequent training program will constitute in part a continuing program of supplementary selection. Basic selection in recruiting of volunteers and psychiatric screening will point to sound high-level personnel motivation and qualification of each individual astronaut.

Progressive personnel selection and elimination in screening will most likely be based on selective proficiency measures and standards accruing as a result of the crew training regime established for APOLLO.

It is apparent that a larger number of prospective crew members will be required as compared to the Mercury program. Over several years as many as 100 crew members may be necessary.

The criteria for selection for Mercury included:

1. Age - under 40
2. Height - under 71 inches
3. Physical condition as defined by standard Navy-Air Force aviation personnel health criteria

4. Bachelor's degree in a natural science
5. 1500 hours flying time
6. Graduate of Air Force or Navy test pilot school
7. Qualified jet pilot

These requirements resulted in a manpower pool of slightly over 100 men. The reduction of this initial pool to 32 men resulted in part from an examination of medical records. Elaborate stress-testing further reduced this group to the present 7 members. Perhaps the most serious restriction on the selection process was the requirement for extensive flight and test pilot experience.

4.2 BIOMEDICAL CRITERIA

Biomedical criteria enter the complex selection process in accordance with the environmental stresses to which the crew will be exposed. Such criteria must of necessity be based upon the Mercury experience, both the techniques for initial selection and the subsequent validation of these techniques in operational Mercury flights.

Some important points of this selection program are:

4.2.1 Metabolic Requirements

An essential quality for the astronaut is stamina. The athlete is able to maintain a relatively steady physiological state with physical exercise because he is able to augment the supply of oxygen required for increased metabolism. For a given amount of work, the athlete has a smaller oxygen consumption than the non-athlete. The athlete will tolerate low oxygen partial pressures longer than the non-athlete. Therefore, the crew should probably be moderately active athletes.

4.2.2 Smoking

Because of the problems of CO disposal and oxygen depletion, the crew should be nonsmokers for at least a year prior to the mission. Carbon monoxide produced in smoking competes with oxygen for a place in the hemoglobin molecule (McFarland, 1953).

At body temperature, the affinity of hemoglobin for CO is about 210 times as great as its affinity for O_2. Both cannot be carried by the same hemoglobin molecule at the same time.

4.2.3 Temperature and Humidity

Individual tolerances to extreme conditions of temperatures and humidity vary. A selection criterion may therefore consist of the ability to solve problems and make decisions under these extreme conditions. In Operation Manhigh, McClure suffered extraordinarily high ambient temperature and humidity, which eventually raised his body temperature to 108.5. He was still able to operate the craft, solve problems, and make rational decisions. Most men would have suffered a catastrophic physiological collapse under those conditions.

4.2.4 High G Loads

Individuals vary in their ability to withstand high accelerative forces. Such tests may serve further as criteria for eliminating or selecting candidates.

4.2.5 Psychophysiological Stamina

In addition to the obvious requirement that the crew be in top physical condition, the crew must possess psychophysiological stamina (Simon, 1960). Crewmen should be selected for their response to emergency stresses of this type. They require a combination of deep physical reserves plus determination to use these reserves. One of the indicators for these capacities seems to be the ability to relax under stress, to sleep when appropriate, and to mobilize immediately, when rapid action is demanded. Selection for tolerance for high g, cardiovascular efficiency, bends, and cabin pressure variations should be considered.

There are several important psychophysiologic factors for which training and habituation, in addition to selection factors, are essential. Notable among these are diet, adaptation to in-cabin rations and conditions of eating and drinking, deprivation of alcohol, sex, recreation, normal social stimuli, and the factors of confinement and accumulated strain.

One selection method may be mentioned for research possibilities. Because motion sickness may represent a threat and is an important element in ground-based training devices, attention should be given to cupulometry as a means of selecting individuals resistant to motion sickness. Cupulometry consists of placing a subject in a rotating chair, darkening the room and bringing the subject to a desired velocity of rotation at a "subliminal" rate of acceleration. Once the desired velocity is reached, the subject is rotated for a short time to allow the cupula to return to its normal position. The chair is then brought to a smooth but rapid stop within one to two seconds. This process is repeated for a number of different velocities, for the most part between 5 and 60 degrees per second. After each trial, the duration of the ocular nystagmus and the duration of postrotational sensations are measured. Nystagmus cupulograms and sensation cupulograms are produced by plotting the logarithim of the velocity versus the duration of the nystagmus or sensation. DeWit has indicated that the slope of the cupulogram is primarily a function of habituation to vestibular stimuli, the slope becoming less steep as the subject becomes more accustomed to the stimulus. Motion sickness tendencies could thus be isolated. Studies in this country (Mann and Cinella) have only partially verified the DeWit work and more study is needed.

Biomedical selection criteria such as described above must be re-examined following the flight experience of the Mercury astonauts in order to determine where requirements as indicated above can be relaxed in order to expand the available manpower pool. Relaxation of such criteria may lead to more extensive training requirements of subjects with the skills required for APOLLO. The results of current Astronaut training must be systematically studied in the APOLLO mission.

The volunteers for APOLLO will undergo an extraordinarily rigorous physical examination. The basis for this will be the rather complete selection exam which has been developed in connection with Project Mercury and is described by Schwichtenberg et al (1959). It includes a complete medical, family, and personal history, clinical procedures, laboratory tests, and results of eleven stress tests.

4.3 PSYCHOLOGICAL FACTORS.

Technical requirements for crew members will be largely based on the information, special-skills and psycho-physiological demands of the mission, such as vehicle control, astro-navigation, medical supervision, lunar science, and special reconnaissance objectives. Basic selection criteria will involve jet flight, or rocket flight experience, and hopefully, orbiting time.

The crew should be devoted to research as well as to survival. They will be required to absorb at least a modicum of scientific knowledge in a large number of areas including space medicine and biology, astronomy, physics, and mathematics. Personal history, as well as academic aptitude tests will help here.

Various crew members will need (a) proficiency and experience as pilots of high-speed, high performance jet and rocket aircraft, (b) detailed engineering understanding of the operation and maintenance of the power plant, controls and environment-conditioning apparatus, (c) skill in the assessment of human performance and functioning, as well as survival techniques, (d) detailed understanding of the operation and maintenance of all communications and scientific observation equipment, (e) detailed understanding of the mission plan in relation to navigational and astronomic frames of reference. Fixing initial requirements for most of these proficiences would greatly reduce the selection problem. Aptitude tests, crew selections and flight check techniques based on flying records are available as supplementary techniques.

Little is known at present about how to select for adaptability to the experiences of prolonged weightlessness, sleep-deprivation, boredom, fatigue, confinement, cumulative annoyance, isolation, or the breakaway phenomenon. For that matter, whether or not many of these, especially the lattermost two, are real and applicable aspects of the APOLLO mission is highly speculative at the present time. At any rate, actual exposure to these stresses will produce fall-out in the initial selection group. Shart and his colleagues, Stunkel et al, and Eilbert and Glaser found evidence that biographical data, self-appraisal, job attitudes, anxiety about interpersonal relationships, job proficiency, peer ratings, and medical records held considerable promise for predicting adjustment to arctic isolation. In general, individuals who adjusted well to arctic isolation were ones who adjusted well to military assignments elsewhere.

Little is known about the selection and training of individuals for vigilance tasks. Lubin and Kleitman have suggested that subjects with strong diurnal temperature curves will perform especially well during their peaks. Wilkinson has shown that subjects who during baseline testing, develop high muscle potentials while operating a serial reaction-time test are less susceptible to acute sleep loss. Brengelmann has evidence that personality variables like intelligence and "rigidity" are positively correlated with good vigilance performance. Bakan and Broadbent both report complex but repeatable positive relations between introversion (assessed with psychometric tests) and good vigilance performance.

Several investigators have reported that in vigilance trials <u>some</u> observers maintain a high performance level throughout the watch, where others suffer a considerable loss in proficiency. If such individual differences are consistent as indicated by the work of Knaff and Pollack, then the performance battery suggested in the midterm report might be used as a selection device to choose individuals who can sustain performance over long time periods.

Selection for good communication is essential. The astronauts need to be good senders and receivers of voice communication through noise and static. The ability to talk intelligibly has been found to have little correlation with the ability to listen in noise. Personnel should be selected on the basis of both listening and talking tests. A well-known testing device is the auditory test no. 8 prepared by the Harvard Psycho-Acoustic Laboratory by Karlin, Abrams and Sanford. It is an IBM-scored multiple-choice listener test consisting of speech recorded against noise. Talking tests have been devised which are administered to one individual at a time by trained judges. Skill in talking and listening improves with practice, training in use of equipment and circuit discipline.

Selection for navigational skill might be accomplished with current Air Force tests.

In addition to such physical and psychological test criteria as discussed, the criterion of inter-personal compatibility and cooperation should be included in APOLLO crew selection. Detailed APOLLO crew job analyses must be accomplished further to highlight and formulate the initial overall and inter-related test criteria for crew selection. The crews must be cross-trained so that each can substitute for the other. Crew cohesiveness will be achieved by selection of compatible and stable personalities, joint training and structuring along military lines.

The astronauts must be selected early in the development of the program and thoroughly committed to its scientific goals. An essential requirement is that they be good at public relations. They must be willing and able to withstand for several years public scrunity of their private lives, their families, their attitudes and goals; this, without becoming prima donnas, for to the very end they will be expected to participate fully in rigorous training, stress experiments and dry-runs.

As mentioned above almost any selection program will be compared to that employed. in the Mercury program, and rightfully so, since a great deal of highly significant information was gathered for Mercury. There are several important points which grow out of such a comparison.

The APOLLO project has certain very different aims from that of Mercury. More than one man will occupy the vehicle and thus the opportunity presents itself to utilize men with a greater variety of backgrounds and proficiencies. The extensive training that will be required may be reduced by selection of some crew members with specific scientific training already accomplished.

It seems apparent that a larger initial pool of personnel will be required since the total program calls for a number of crews. It is important to note that a group of only 110 individuals passed the initial Mercury selection process and of this group 32 were given further tests.

The size of the available population must be weighed against the imposed selection criteria. APOLLO crew members will have more functions and these can be translated into additional selection criteria. At some point, these criteria may have to be reduced in number to provide enough candidates for training. For example, the Mercury selection criterion of graduation from flight test pilot school may be changed to a training requirement for the first APOLLO crews. For subsequent APOLLO crews, this criterion may give way entirely.

5.0 Training

5.1 INTRODUCTION

The analytical data in a training-program outline must include a system description, with preliminary man-machine functions allocation, together with descriptions of environmental conditions, under which the crew must survive and operate. Preliminary estimates must also be made of required vehicle-crew skills and knowledge, and special physical and psychological qualifications. This material is described in the previous sections of this chapter.

The complete personnel support complex must be considered. The broad categories of personnel who perform in the total APOLLO system may be designated as follows:

(1) Space-vehicle crew.

(2) Flight-supporting ground and recovery crews.

(3) Ground Support Crews; in-flight preparation, checkout, servicing, installation, manufacture and assembly, and launch activities.

(4) Scientific monitoring and advisory personnel.

The training requirements then reduce to a statement of required adaptations, knowledges, and skills under given environmental conditions of operation. Areas where training is required are then established on a rational basis in training functions analysis. Proficiency criteria and validity of simulation must be determined in the course of the training program as actual operational data become available, and as experienced astronauts can be relied upon for valid judgements and opinions.

The core-training curriculum for APOLLO must be coordinated with other parallel manned-orbiting satellite programs. Special simulator design required for APOLLO will be based on training functions data and a trade-off between cost and psychological fidelity as they contribute to training effectiveness and attainment of the requisite performance levels. Special training problems in APOLLO ground environment trainers include such things as: (1) Provisions for "realism" in simulation of the astronautical visual scene, (2) identification of, and provisioning for significant operational cues for training, such as acceleration sensations, (3) establishing of significant proficiency criteria for selection and performance assessment, and (4) establishing of integrated crew training requirements for vehicle crew interactions, ground-to-vehicle crew interactions, and global-tracking crew interactions.

5.2 GROUND CREW TRAINING

5.2.1 General Personnel

The overall APOLLO personnel subsystem complex will involve those ground-based teams supporting the mission while the vehicle is in lunar flight, and the ground personnel concerned with landing, search and vehicle recovery after touchdown. The complex will also include the factory-to-launch technician crews in fabrication, transportation, installation, assembly and erection of all booster and vehicle components, including support of the flight crew in preparation for, and execution of launch. For APOLLO and similar manned space flight operations, a requirement for several varieties of scientific specialists may be expected, with these personnel to act in a monitoring, consulting, and advisory capacity or in medical care of the astronauts.

By and large, the ground crews supporting the APOLLO operation will most likely be selected on the basis of an extensive background in missile and space-flight support operations. Special training will largely be acquired in on-the-job situations in earlier and similar flight programs, such as that of Mercury. Special APOLLO training requirements will be spelled out on the basis of required knowledge and skills important to achieving reliability and proficiency, as established in training functions analysis. Training programs and equipments, such as data-flow trainers, visual aids and simulators will be established in this way. The discussion that follows covers only a broad outline of training requirements as conceived on a rational basis. The several categories of personnel for whom training requirements exist include:

a. Key personnel required on the ground to support, control and recover, as necessary, the APOLLO vehicle and crew.

b. Key ground personnel responsible for equipment in the factory-to-launch sequences.

c. Key ground personnel with scientific and medical responsibility.

d. The APOLLO vehicle crew.

5.2.2 Ground Support and Launch Crew Training

It has been found necessary in the past, from the standpoint of missile system reliability, to embrace the ground support complex in a total system concept. This is desirable since every facet of operation from manufacture to erection and launch is then analytically controlled. Thus, a systematic training program may be instituted in important areas that will contribute to proficiency of the human operator and reliability in his performance. For a manned space-flight operation such as APOLLO this is of fundamental importance, since human-operator error in any of the ground support phases could compromise safety of the crew and success of the mission.

Figure II-5-1 presents a representative block diagram outlining the broad areas in a sequence of APOLLO operations leading to launch. The Saturn booster units will be assembled after manufacturing, test fired, cleaned up and prepared for transportation to the staging area. Likewise, the APOLLO vehicle modules will be fabricated, assembled and subjected to necessary environmental and functional tests, incorporating installation of subsystems components with tests and inspection procedure, packed and shipped to the staging area. Integration, maintenance, testing and checkout operations will be completed as necessary and all equipment will be shipped thence to the launching site in final preparation for the mission and launching.

Within this broad framework of processes, a detailed breakdown of the factory-to-launch operations becomes necessary in order to clarify special skills and knowledges required of the human operators.

Sound human engineering in equipment design for testing and maintainability may obviate many training functions in other than on-the-job situations. However, a thorough and detailed training-functions analysis must be completed in order to establish those areas requiring special attention with regard to ground crew proficiency

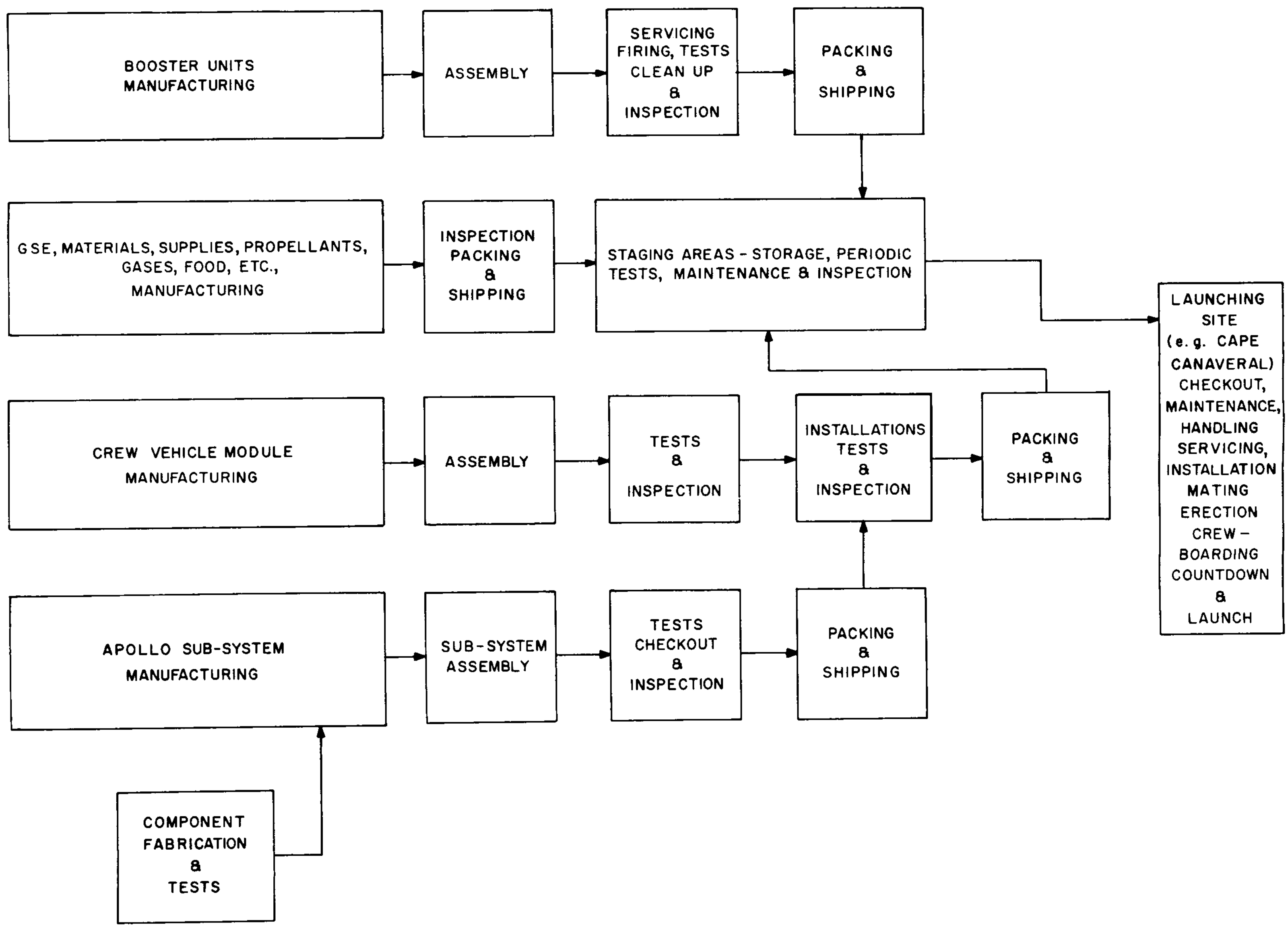

Figure II-5-1. APOLLO manufacturing-to-launch sequences and reliability of operation. This may involve instructions such as given in factory training programs for launch crews, the use of special data-flow trainers and mobile training units for use at staging areas and launch sites, and special instruction in diagnosis of malfunctions and corrective action, and in use of special/automatic checkout equipment and theory and maintenance thereof.

Items of training equipment and aids will be incorporated in comprehensive ground-crew training programs as follows:

a. Booster unit diagrams and functional data flow trainers.

b. Animated panel trainers covering the vehicle system and subsystems.

c. Technical training charts.

d. Data-flow trainers on the vehicle avionics and automatic checkout equipment.

e. Technical information bulletins on booster units and vehicle systems.

f. Ground-cooperating systems procedural trainer requiring complex individual and crew tasks in testing.

g. Launch-control procedural trainers.

Complete analysis of system activities and the plan of operation must be completed in order to establish the complex of manual and intellectual skills required in support of the APOLLO system, and to provide the necessary information, instruction and training curricula essential at the outset as well as throughout project development and operation.

5.2.3 Flight Supporting Ground and Recovery Crew Training

The tracking networks necessary to support such space vehicles as APOLLO will require highly complex instrumentation, optical, and transmission equipment on a global basis. Recovery-crew activities must also be coordinated in the landing-approach and touchdown sequences, or the air-sea-ground search and retrieval operations.

The intent in design of the APOLLO-vehicle crew and ground-supporting flight crew functions is for optimal allocation of tasks. Certain "normal" and "emergency" functions, as will be established, will be most efficiently accomplished by the vehicle crew. These functions will normally include such activities as navigation, primary vehicle control, reconnaissance, and general mission control.

Certain normal and emergency functions may be allocated to the ground-based command centers, such as backup tracking and navigation, corrective action for telemetry or instrumentation failure.

Manning requirements of the command center station(s) for APOLLO operations will include: (1) skilled ground and system-specialist control operators, (2) physicians, (3) lunar scientists, and (4) station engineers and technicians.

Special training will be necessary for the complex skills required in manning the ground-station consoles for such functions as:

(1) Coordination with related tracking stations for continuous global tracking of APOLLO.

(2) Normal trajectory computations, tracking, monitoring and transmission to the APOLLO vehicle.

(3) Advising on secondary mission functions.

(4) Monitoring vehicle subsystem status, including physiological condition of crew.

(5) Directing repair, modification or alternate use of on-board equipment.

(6) Coordinating position data with recovery crews.

A training program for the ground-control operators and system specialists will include such training functions as follows:

(1) Indoctrination on the APOLLO system.

(2) Familiarization with pertinent details of the APOLLO mission, such as programmed trajectories, and individual tasks of the vehicle crew members.

(3) On-the-job training in coordination with tracking stations and, as necessary, recovery facilities.

(4) Practice in interpretation of radio display signals for reading of vehicle and mission status, diagnosis, trouble shooting and corrective action.

(5) Practice in voice communication with the vehicle crew at the ground-environment simulation facility.

Recovery crews will be required to accomplish search and retrieval of the vehicle in a semi-ballistic configuration, or an emergency landing of the glide-vehicle configuration. This will entail preparedness to recover, at normal landing sites, on the sea, and various geographic areas of desert, as well as jungle or arctic regions. Various land, air and sea crews, such as employed in the U.S. Navy's First Fleet in the Pacific Missile Range may require special drill for search and recovery of the APOLLO vehicle.

5.2.4 Scientific and Advisory Personnel Selection and Training

The areas in which scientific and advisory personnel will require specialization will include various medical specialties, and those areas most pertinent to the vehicle system and mission. APOLLO system and design engineers will be on hand as equipment specialists, who, in addition to being fundamental experts in the design and operation of the system on a project basis, will have undergone indoctrination on crew problems. Other advisory personnel would include astrophysicists, geologists and geophysicists specializing in lunar topography, radio-biologists, geneticists, et. al., all of whom will have undergone indoctrination on the APOLLO system and mission objectives.

Medical specialists will work as a team at the tracking station(s) with engineers who will check APOLLO compartment environments, while the medical monitoring would include such things as heart action, respiration and body temperature. In addition, physicians will be assigned to various recovery teams to administer medical treatment to returning astronauts as required.

Engineering personnel must, of course, keep abreast of the state-of-the-art throughout APOLLO project developments. Special symposia and instruction will be required to familiarize them with particular crew and environment problems.

Current state-of-the-art, and integration of space-probe and space-flight experience of such parameters as radiation, meteoritic hazards and solar storms, must be maintained by assigned astrophysicists. This may be accomplished by such means as lectures, symposia, and dissemination of data in report form. Current lunar data may also be disseminated in similar fashion.

Space surgeon training may consist of a general background provided in the tri-service pool of medical specialists that is currently developing at Cape Canaveral under the Air Force Medical Training Center Command.

Medical personnel assigned to monitoring functions on the Command-Center tracking team will undergo detailed training on the APOLLO system, including medical records and histories of each individual APOLLO astronaut. On the other hand, for those assigned to recovery teams, only informal familiarization with the APOLLO project may be necessary.

5.2.5 Complete Systems Training

A complete simulation network may be conceived for complete systems simulation. System problems may be run and scored for ground training purposes, to develop procedures and spot weaknesses in advance of the actual APOLLO mission. Special filming and computer programming simulation may be employed at each station, in communication with all other stations, involving the complete space-to-ground loop and ground crew complex for integrated systems training.

5.3 FLIGHT CREW TRAINING

5.3.1 Training Considerations

Man is scheduled to play an active part in the operation and control of space vehicles because the use of his unique capabilities will increase both the scope and reliability of space missions. Man has superior ability to recognize patterns, learn from experience, reason inductively, utilize a large and well-organized memory, adapt to changing conditions and unforeseen circumstances, and time-share his many functions. These capabilities far exceed those possible with equipment occupying the same volume and weighing the same as man, plus the environmental devices required for his safety and comfort. Thus, the presence of man as an active participant in a space vehicle increases the scope of space missions by exploiting man's versatility. The reliability of the vehicle and the probability of its mission being a successful one are increased because man is capable of diagnosing and correcting equipment malfunctions and coping with emergency conditions while the vehicle is in operation.

In order to exploit these talents fully, comprehensive training will be necessary not only to utilize man's capabilities, but to compensate for his limitations under the conditions encountered in space flight. The nature of these conditions and of the stresses affecting performance have been considered elsewhere in the body of this Report.

In developing a training program for APOLLO crewmen, reasonable selection procedures, in conjunction with a low selection ratio (the selection ratio is the number chosen divided by the number to be chosen from), will yield a group of trainees of high initial skills as well as high motivation. This is fortunate, since the performance requirements of APOLLO astronauts are quite rigorous.

To enable trainees to reach required performance standards, a clear definition of training requirements is needed. These requirements are stated in implicit form elsewhere in this Report. To determine whether trainees have, in fact, achieved these standards, a comprehensive performance evaluation program is required. This program is essentially one of quality control; in addition to serving as a check on the final product of training, it is also useful for raising trainee motivation, improving learning by providing useful feedback, and indicating in objective terms the exact state of the training program.

5.3.2 Training Program

The proposed training program is outlined in Table II-5-I. For trainees without prior space training, an eighteen-month program is required. Modification of this program would be required for men who have flown Mercury or DynaSoar missions; for astronauts flying more than one mission comparatively brief refresher training will be required between missions. The proposed utilization of trained personnel for several missions is described elsewhere.

The hours required for physical training and for maintenance of proficiency are quite sensitive to changes in the duration of the training program. The eighteen-month period includes a small time cushion for training needs not foreseen at this time; it is included in the time allocated to maintenance of proficiency and "keep-current" training. No time is allocated for maintenance of proficiency in flying conventional aircraft.

Differentiation of training for the three crew positions is not shown, although it may be desirable. The exact nature of the training program for each of the three crewmen depends upon the number of positions each must be qualified to fill, and the level of qualification for each. A somewhat shorter training cycle would result from having each man specialize in one or two positions, rather than being fully competent in all three.

Physical training will best be accomplished by closely supervised gymnastics (push-ups, etc.) rather than by loosely supervised or unsupervised sports (e.g., golf, baseball). Sports will be included in the program, to the extent time permits, only after the required degree of physical fitness has been achieved.

Changes in equipment and procedures occurring after training but before launch, will require a continual updating of training. This supplementary training will be

[redacted]

TABLE II-5-I. VEHICLE CREW TRAINING PROGRAM (PRELIMINARY)

Training Requirement	Hours	Training Methods	Training Devices	Means of Obtaining Criteria of Proficiency
1. Knowledge and Concepts				
A. Space Fundamentals, e.g. Celestial Mechanics Space Navigation Propulsion Human Physiology First Aid Survival Geography Electronics	350	Lecture-Demonstration Automated Instruction Individual Study	Slides Motion Pictures Mock-Ups Teaching Machines Texts	Paper-and-pencil tests Teaching machine records
B. APOLLO Subsystems (theory and method of operation, detection and correction of malfunctions) Life Support Electric Power Communications Flight Path Management Propulsion Scientific Observation	400	Lecture-Demonstration Individual Study Automated Instruction Discussion	Visual Aids Mock-Ups Teaching Machines Manuals Operational Equipment Part-task trainers	Paper-and-pencil tests Teaching machine records Job-sample tests
2. Adaptation and Physical Conditioning				
A. Physical Training	350	Individual Exercises Competitive Sports	Gym Equipment Sports Equipment	Physical Fitness tests
B. Adaptation to Stresses of Space Environment, e.g. Acceleration Weightlessness Tumbling Heat	250	Exposure to stresses	Open loop centrifuge Tumbling Devices Vibration Devices Water Tank C-130 & KC-135 Zero-G flights Mercury & Dynasoar flights	Physiological Measures Performance Tests

[redacted]

TABLE II-5-I. VEHICLE CREW TRAINING PROGRAM (PRELIMINARY) (Continued)

Training Requirement	Hours	Training Methods	Training Devices	Means of Obtaining Criteria of Proficiency
3. Individual Skills				
A. Mission Phase Procedures for Pre-launch Launch Boost Midcourse Outbound Lunar Orbit, Reconnaissance, and Initiation of Earth-bound trajectory Midcourse Inbound Re-entry Landing Survival	500	Lecture-Demonstration Individual Study Automated Instruction Controlled Practice	Visual Aids Manuals Teaching Machines Part-task trainers Mission Module Simulator Re-entry Module Simulator Navigator Station Simulator-centrifuge combination	Paper-and-pencil tests Teaching Machine Records Job Sample Tests
B. In-Flight Maintenance	250	Lecture-Demonstration Controlled Practice Automated Instruction	Visual Aids Mock-Ups Operational Equipment Teaching Machines Maintenance Trainer Simulators (programmed malfunctions)	Paper-and-pencil tests Teaching Machine Records Job Sample Tests
4. Crew Coordination Skills				
A. Integrating Mission Phases	150	Controlled Practices	Mission Module Sim. Re-entry Module Sim. Navigation Station Sim.	Job Sample Tests

[redacted]

TABLE II-5-I. VEHICLE CREW TRAINING PROGRAM (PRELIMINARY) (Continued)

Training Requirement	Hours	Training Methods	Training Devices	Means of Obtaining Criteria of Proficiency
4. (cont)				
B. Coordination among Crew Members	150	Controlled Practice	Mission Module Sim. Re-entry Module Sim.	Job Sample Tests
5. Refresher, Retention, and Keep Current Training	600	All of above, as appropriate	All of above, as appropriate	All of above, as appropriate
TOTAL HOURS (18 month program)	3000			

[redacted]

accompanied by the same thorough evaluation procedures as the original training, to assure that the astronauts know well the then-current system.

Automated instruction by means of teaching machines or scrambled textbooks is emphasized because its use:

a. Forces the instructional staff to conceptualize clearly each element that must be learned.

b. Permits close control of learning, thus allowing modifications to follow closely changes in equipments or procedures.

c. Provides concurrent evaluation of trainee learning, closing the loop of the training process by providing knowledge of results.

5.3.3 Training Equipment Requirements

A large number of equipments will be needed for training APOLLO astronauts to the required degree of proficiency. These are listed in Table II-5-II. The most expensive of the special training devices and the one with the longest lead time is the Mission Simulator. The Mission Simulator consists of the Mission Module Simulator and Re-entry Module Simulator, the two component units being usable together or separately. Of substantial complexity also is the Vehicle Commander's Station Simulator, to be usable, closed loop, on the AMAL (or equivalent) centrifuge.

A. GENERAL REQUIREMENTS FOR THE APOLLO MISSION SIMULATOR

In order to meet the training requirements described above, the following four general requirements must be met for the Mission Simulator, and, to a large degree, by the Vehicle Commander's Station Simulator.

1) Stimulus Conditions. These must duplicate reasonably closely those of the space vehicle to which generalizations are to be made.

2) Response Capabilities. The trainees in the simulator must be capable of responding to these stimuli in the same manner in which they would in the space vehicle.

TABLE II-5-II. APOLLO VEHICLE CREW TRAINING EQUIPMENTS

1. Simulators

   A. Mission Module Simulator
   B. Re-entry Module Simulator

   May be used separately or together: When used together, they constitute the Mission Simulator.

   C. Vehicle Commander's Station Simulator (to be usable, closed loop, on human centrifuge)

2. Adaptation Devices

   A. Open Loop Human Centrifuge
   B. KC-130 and KC-135 (for zero G Keplerian trajectory flights)
   C. Water Tanks (for zero G simulation)
   D. Air Bearing Platform
   E. Tumbling-Disorientation Device
   F. Vibration-Buffeting Device
   G. Confinement-Life Support Equipment Trainer

3. Part-Task and Procedures Trainers - including:

   Antenna Steering Training
   Solar Collector Orientation Trainer
   Vehicle Egress and Survival Trainer (land)
   Vehicle Egress and Survival Trainer (water)
   Manual Navigation Procedures Trainer
   Emergency Procedures Trainer(s)
   In-Flight Maintenance Trainer(s)
   Scientific Observations Trainer

4. Visual Aids and Mock-ups

5. Teaching Machines

3) Performance Evaluation. The behavior or performance of the trainees must be evaluated precisely, comprehensively, and rapidly.

4) Programming. It must be possible not only to modify the displays and response mechanisms rapidly between sessions, but also to modify the aspects of the subject's performance that are measured. Therefore, very flexible programming of the facility is required.

No ground facility can furnish an environment completely duplicating that encountered by the crew of a space vehicle. Aside from limitations in the state-of-the-art, there are limitations imposed by the cost factor. The cost of simulation increases more and more rapidly as complete duplication of the space environment is approached. For this reason, a compromise must be made between fidelity of simulation and cost. Because errors resulting from fidelity limitations are likely to be expensive (in dollars, in degradation of mission capability, and perhaps even in lives), a rather high degree of fidelity must be secured with the APOLLO Mission Simulator, even though considerable expense may be involved. Trade-offs will be required in areas where technical or technological limitations make complete simulation impossible or prohibitively expensive; the simulation of sub- and super-gravity conditions are cases in point.

B. CAPSULE ENVIRONMENT

In order to perform the required training, the APOLLO Mission Simulator must contain a capsule together with ancillary equipment necessary to provide the trainees with appropriate stimulus conditions and response capabilities. Among the more important of the features of the capsule environment are the following:

1) Configuration. The interior layout of the simulator capsule should be a virtual duplicate of the space vehicle capsule. The location and arrangement of "furniture", controls, displays, and life support equipments should all be considered. Since simulation of the exterior of the capsule is unimportant for most training purposes, the exterior should be designed not so much for realism as for easy modification of the interior so that different equipments and/or layouts can be used. It is probable that, to attain needed flexibility and high utilization of the facility, more than one capsule will be required;

the particular capsule required would be "plugged in", while the alternate(s) is (are) undergoing modification for later use.

2) Displays. All displays relevant to a given test, including displays competing for a subject's time or being looked at by him for any purpose, should be duplicated. This duplication should apply not only to the form of the display, but also to the characteristics of the information provided. For example, the lags, accuracies, and other idiosyncracies of particular instruments should be reproduced.

   A special display problem occurs, since it is planned to provide the astronauts with a view of the extra-capsule environment. Such a visual field can be achieved by direct vision through a viewing port or by means of a periscope. In any case, the extra-capsule visual field should be simulated. This requirement implies the normal capability of a planetarium plus the simulation of sun, earth, and Moon.

3) Controls. All controls present in the vehicle that are relevant to a given test should also be present in the capsule. It is of importance not only to duplicate the appearance and feel of each control, but also its effect upon various displays.

4) Life Support Equipment. For life support equipments with which the astronaut has direct contact, such as pressure suits, food containers, etc., the actual equipment should be utilized. Certain portions of the life support systems may be simulated without prejudicing the results of many tests; for example, the oxygen supply itself might be simulated, as long as the controls associated with it produce realistic results.

5) Motion. The acceleration encountered by an astronaut in the course of a space mission constitute an important part of the astronaut's environment. These accelerations are important in two ways. First, they furnish a context for actions that may either facilitate or hinder a given action. Second, accelerations may furnish kinesthetic cues as to vehicle performance.

For purposes of this discussion, accelerations may be classified into four categories:

a) Subgravity. Since sustained weightlessness cannot be reproduced in a ground facility, it does not have to be considered as a feature of this facility, desirable though it might be.

b) Super-gravity. The accelerations to be encountered during launch and re-entry may be on the order of 8 G. Sustained accelerations of this magnitude can be produced only on a centrifuge, and for this reason the simulator computational facility should be capable of tying-in with the centrifuge, the gondola of which may serve as a suitable capsule. The only alternate to this arrangement is the building of a centrifuge as an integral part of the simulator, which would be extremely costly and would not furnish an appreciably better capability.

c) Buffeting and Vibration. These accelerations can be produced with conventional flight simulator motion equipment for the most part. During a mission, buffeting and vibration are of most importance during boost and re-entry, and hence subject to sustained multi-G acclerations. Therefore, it appears sensible to provide buffeting and vibration only in conjunction with the sustained multi-G accelerations of the centrifuge.

d) Accelerations during Landing Approach. These are the typical accelerations that would prevail for a perfectly straight approach in calm air. Because of air turbulence, roll, bank, and pitch movements, and vehicle accelerations and decelerations, the magnitude of the acceleration vector can be either greater than or less than 1 G, and its direction can be other than from the astronaut toward the floor of the capsule. These accelerations often furnish pilots with valuable cues that aid them in controlling aircraft, because both instrument lags and instrument interpretation time are virtually eliminated.

However, a recently completed study at Link indicates that the task of flying an ILS approach and night landing on an ME-1 simulator is more difficult with motion than without it, subjects making greater lateral error, although less glide-path error.

Because the functions of these acceleration cues, both in providing useful information and in leading pilots astray, are not known in detail for space vehicles, it appears desirable that a motion system incorporating these cues be provided in the simulator. It is likely that the control of the space vehicle during approach and landing will be quite critical. Thus, even though the motion system would be used during only a small fraction of the mission, this fraction is a most important one.

Although this motion system will not replace the centrifuge for adaptation to the effects of buffeting and vibration, the opportunity to produce these effects, without tying up the centrifuge should be of value.

6) Mission Duration. The simulator should be capable of running 14-day missions, maintaining the integrity of simulation for the entire period. Implied here are:

   a) High reliability of simulator systems;

   b) A maintenance philosophy that permits necessary simulator maintenance to be accomplished during a mission or experiment;

   c) Instructor facilities adequate for around-the-clock use; and

   d) Extensive data gathering, data storage, and evaluation facilities.

C. DATA PROCESSING AND DISPLAY

The essential products of the tests, exercises, or studies to be conducted on the simulator are the data relating astronaut performance to the conditions or stimuli under which the performance takes place. The timely availability, form, and comprehensiveness of these data are of primary importance to the instructor.

The flow of data of interest to the instructor is shown in Figure II-5-2. Five kinds of raw data are available:

1) Display readings of vehicle instruments.

2) Positions of vehicle controls, including switches.

[redacted]

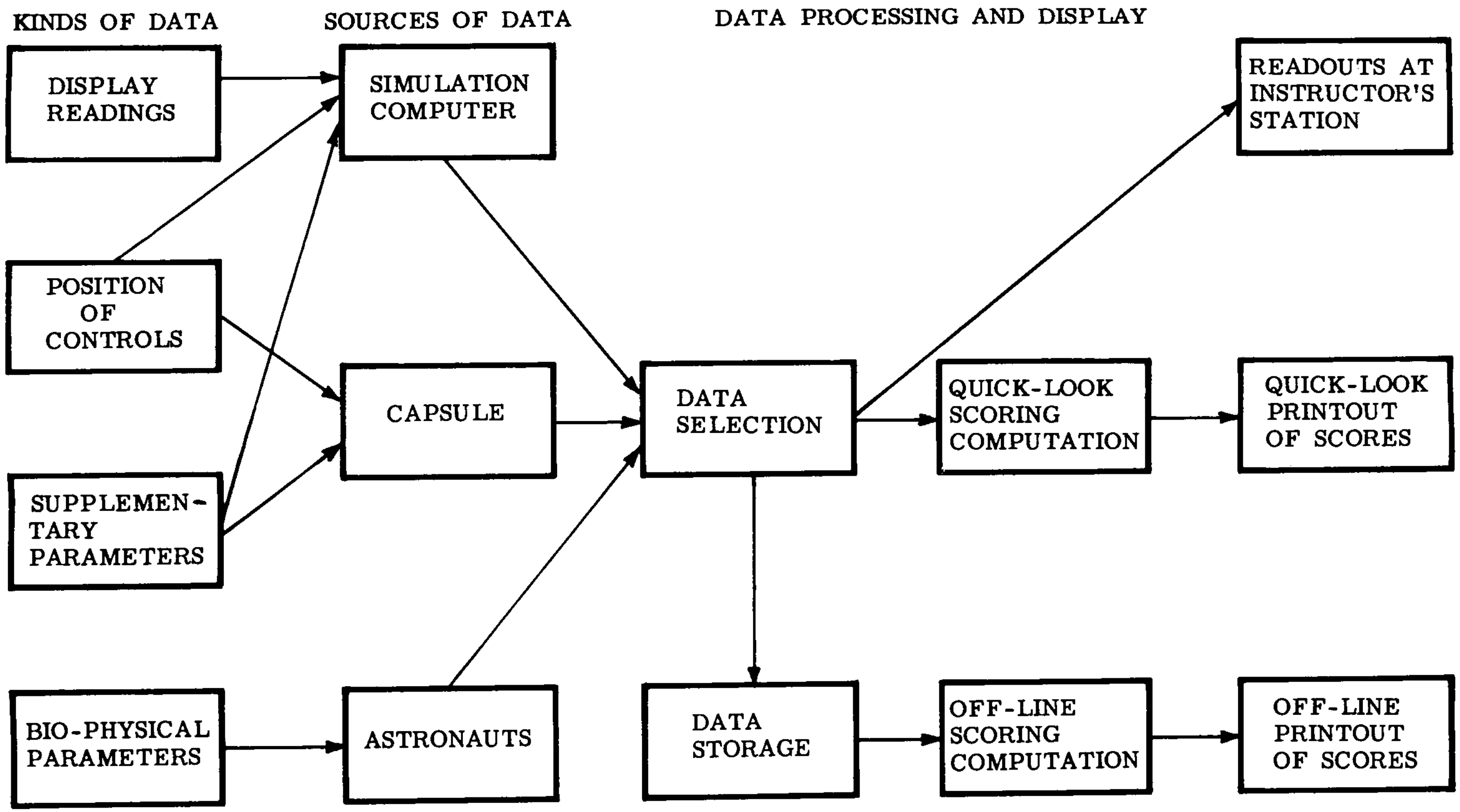

Figure II-5-2. Data flow

[redacted]

3) Values of biomedical parameters, such as pulse rate, that may be used to evaluate the physiological and/or psychological functioning of the astronauts.

4) Verbal reports of the astronauts

5) Values of vehicle parameters, such as capsule temperature and parameters of visual display, not given in 1) or 2) above.

Most of these data can be obtained from the simulator computer. Some data will have to be obtained from the capsule, and the biomedical data will, of course, be obtained from sensors near or on the astronauts' bodies.

A very large amount of data will be generated by the simulator. At least 150 instrument channels will be used in the APOLLO vehicle, with upwards of 200 controls, including switches. To these must be added (for the simulator) biomedical channels (estimated need of at least 20) and miscellaneous capsule sensors (at least 30 channels). The instructor thus has available no less than 400 channels of data — some changing rapidly and others changing no more than once during the mission, some of vital importance and others of much lesser importance.

Information concerning about 10 percent of these channels will be needed by the instructor during a simulated mission. These data would serve to indicate whether the mission is going according to plan, whether the trainees are safe, etc. To provide this information during the training session, three kinds of displays are required. The first is an assembly of "repeater" instruments, furnishing the instructor with the information that is available to the trainees in the capsule, but not necessarily in the same form. It may be possible, in many instances, to use a standard type of meter movement or instrument drive, and, by employing a suitably calibrated dial, keep the instructor's console much the same, even though extensive modifications of capsule instruments take place. It may also be feasible for the instructor to time-share a number of inputs into a smaller number of instruments, even if this is not done in the capsule for reliability or other reasons.

A second kind of display is comprised of the indicators of switch positions and other control positions. These, too, may be time shared, if desirable.

The third kind of display is not a simple readout, but a computation in real time of facets of astronaut functioning, such as, for example, tracking error. Hard copy of these on-line scores is desirable to relieve the instructor of the need for excessive transcription of meter readings to the neglect of more important duties.

Since the conduct of training on the Mission Simulator is not inexpensive, and the decisions based upon them are so important, it is essential to utilize data as effectively as possible. By tape-recording vital data during a training session, it will be possible for the instructor to analyze these data later on a large, general purpose digital computer. If warranted, he can re-analyze the data with a different scoring program, and then modify training appropriately.

Thus, the simulator should employ both on-line and off-line scoring. The on-line and off-line scoring systems complement each other well. The advantages of each of the two can be summarized as follows:

With the on-line system, a limited number of scores would be immediately available to the instructor and (if desired) to the trainees. Thus, any difficulties can be ironed out on the spot. This prompt knowledge of results will also make for improved progress in training.

The off-line system, recording all the data during an exercise can be processed readily by large digital computer, such as the 7090 which is available at General Electric. The computer itself would be required only a few hours per week. This would make possible a sophisticated scoring program, as well as running the same data tape with several different scoring programs to evaluate and to improve the scoring programs themselves. The large computer would be easier to program, because of the availability of more subroutines and more advanced compiler and executive programs.

With the off-line system, virtually all data would be recorded. This has the advantage of furnishing a permanent record of each exercise or run allowing a playback of the mission.

In specifying the equipment needed for the scoring, monitoring, and recording of crew member actions, the following difficulties are faced:

1) Some of the duties or functions to be assigned to the astronauts have not been definitized.

2) Even for those duties or functions which are known to be required of the astronauts, details of the ways in which they will execute the functions are lacking. The form and information content of their displays, the configuration of their controls, their task environment, including other tasks competing for time and attention, and criteria of adequate performance — all these must be specified more precisely.

3) The astronauts' tasks will vary from one vehicle configuration to another and on a given vehicle from one mission to another. Even on the same mission on the same vehicle, a progressive evolution of the astronauts' tasks can be expected as more is learned about vehicle and astronaut performance.

Because of these ambiguities, at this time, scoring equipment or scoring programs cannot be designed around a specific set of required skills and behaviors, as could be done for an operational bomber system for which mission requirements, vehicle characteristics, and crew performance requirements are quite well defined. Instead, the scoring equipment for the Mission Simulator must have flexibility so that it can be used over the entire range of mission requirements, vehicle characteristics, and crew performance characteristics that are reasonably likely to be associated with APOLLO vehicles.

To illustrate the scoring equipment requirements, the scoring of a hypothetical astronaut task will be described.

Hypothetical Task: Emergency Return Due to Excessive Leak: An emergency situation is programmed in the cabin environment subsystem, e.g., one that leaves only a 2-1/2 hour supply of oxygen. The astronaut must first detect the oxygen malfunction, determine alternate modes of action, and then, if abort is indicated, select the most desirable landing area on the earth's surface that falls along the predicted ground track of his orbit. Scorable items for this task might include:

a) Time required to detect emergency

b) Time to evaluate alternatives

c) Soundness of alternate selection

d) Time required to select landing area

e) Soundness or merit of selection of landing area

Assuming that scoring is performed by a general purpose digital computer, the program for scoring these items would resemble the flow diagram shown in Figure II-5-3.

To execute this program, the scoring computer must have such inputs as:

a) Information on the readings of displays available to the astronaut.

b) The positions of the controls and switches the astronaut employs.

c) Values of vehicle parameters not given in a) and b).

d) Readings from the biomedical and physiological instrumentation used to evaluate the physiological and/or psychological condition of the astronaut.

Many of these inputs will be available from the simulator computer; some of these may require transformation into usable form. Others will require special inputs.

D. LAYOUT AND CONFIGURATION

Since accurate data must be obtained during runs of up to 14 days in length, it is important that the facility provide for 1) instructor comfort, and 2) maintenance during operation. A possible configuration of the facility is shown in Figure II-5-4. The aisles between the computer racks allow easy access for maintenance. It is important that the design of the electronics emphasize reliability as well as ease of maintenance.

The instructor's console, located near the capsule, should be a comfortable working place. The console illustrated features panel wrap-around in both horizontal and

SENSE INSERTION OF OXYGEN MALFUNCTION BY INSTRUCTOR OR BY PROGRAM.

DETECT AND STORE RELEVANT VEHICLE INSTRUMENT READINGS.

STORE TIME OF MALFUNCTION INSERTION IN REGISTER A.

ENTER SUBROUTINE THAT WILL RECOGNIZE WHEN EMERGENCY IS DETECTED; AMONG THE INDICATORS OF EMERGENCY DETECTION ARE:

1) MANIPULATION OF OXYGEN CONTROLS
2) MANIPULATION ON RE-ENTRY COMPUTATION PANEL
3) PHYSIOLOGICAL INDICATIONS OF STRESS SUCH AS FASTER PULSE RATE, ETC.

STORE TIME OF MALFUNCTION RECOGNITION IN REGISTER B.

SUBTRACT CONTENTS OF REGISTER A FROM THOSE OF REGISTER B AND PRINT OUT WITH LABEL "OXYGEN MALFUNCTION DETECTION TIME."

ENTER SUBROUTINE TO DETERMINE WHEN LANDING AREA IS SELECTED.

SCORE MERIT OF SELECTION BY EVALUATING IT ON PREDETERMINED AND STORED CRITERIA, OR BY COMPARISON WITH LIST OF POSSIBLE LANDING AREAS, ALSO SCORED, EACH OF WHICH HAS FIGURE OF MERIT ASSIGNED TO IT.

STORE TIME OF SELECTION IN REGISTER C.

SUBTRACT CONTROLS OF REGISTER B FROM THOSE OF REGISTER C AND PRINT OUT WITH LABEL "LANDING AREA SELECTION TIME."

PRINT OUT MERIT OF SELECTION WITH LABEL "MERIT OF SELECTION."

COMBINE SCORES OBTAINED WITH MISSION, PHASE, AND OPERATOR SCORES, AS REQUIRED, AND PRINT OUT.

Figure II-5-3. Fragment of a scoring program

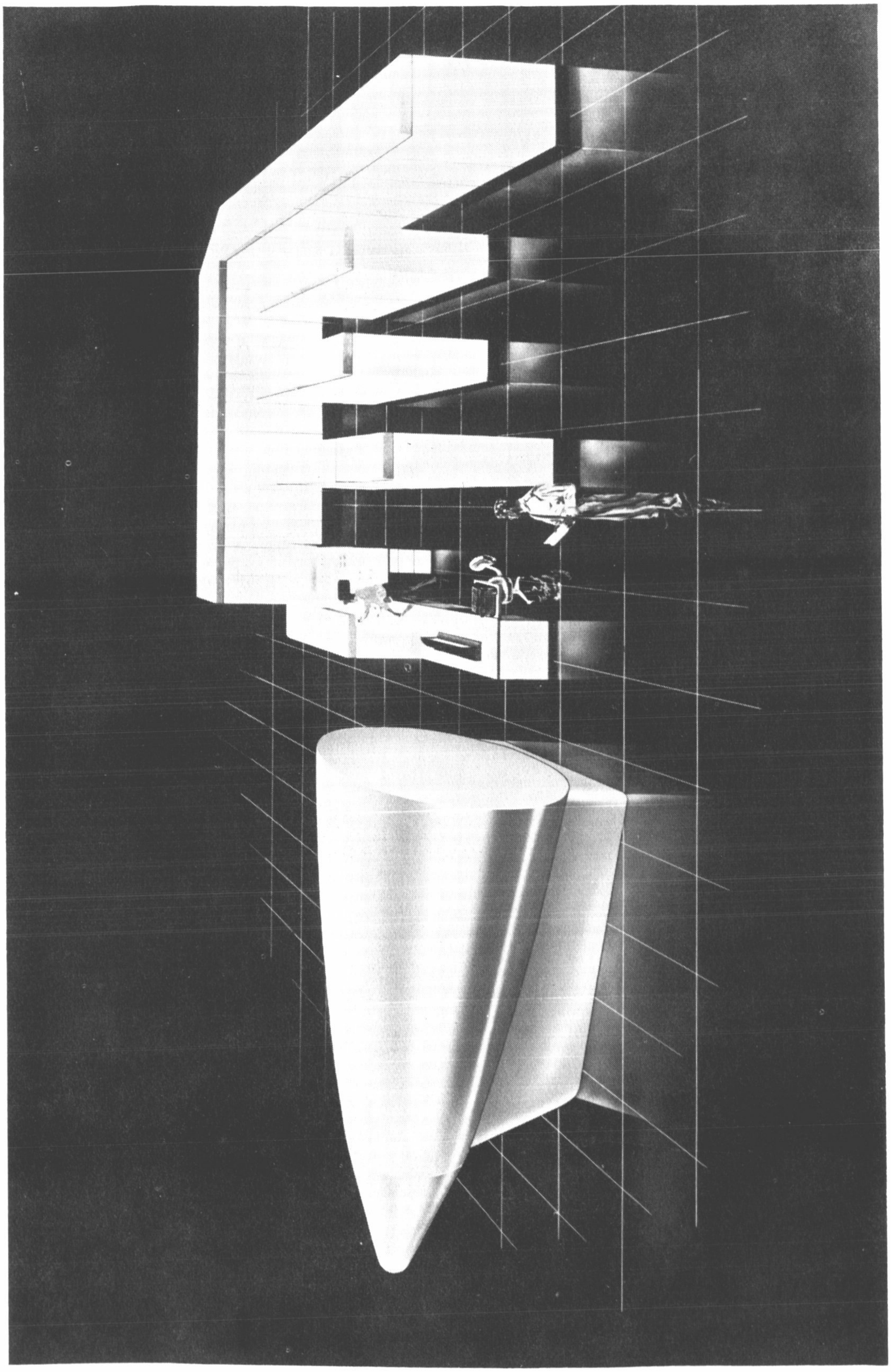

Figure II-5-4. Artist's conception of APOLLO training facility vertical dimensions to assure that controls are within easy reach and displays are viewed orthogonally. It is evenly lit with glare-free lighting, and is designed in accordance with the best human engineering standards. A workspace for two people (e.g., instructor and operator) is provided.

E. PERSONNEL

The following job functions are required for the planning, conduct, and data analysis of exercises. Some of these functions may require more than one person; conversely, one person may be able to perform more than one function.

1) Chief Instructor. Determines exercises or experiments to be run, and exerts over-all supervision of their planning, conduct, and analysis.

2) Programmer of Stimuli. Sets up computer complex and capsule environment to provide proper response-display dependencies and stimuli to the subjects.

3) On-Line Scoring Programmer. Writes and debugs (prior to the exercise) the computer program(s) for on-line scoring.

4) Off-Line Scoring Programmer. Writes and debugs the computer programs for off-line scoring, using information gained from on-line scoring and previous processing of data tape.

5) Medical Monitor. Oversees the physical condition of astronaut subjects by direct observation (through ports in the capsule and/or by closed circuit television) and by monitoring readouts of physiological variables, such as pulse and respiration.

6) Facility Operator. Performs such needed functions during an experiment as changing data tapes, changing variables recorded (using a patch panel), etc. Monitors "repeater" instruments and other displays to assure satisfactory running of experiment.

7) Maintenance Technician. Performs required preventative and corrective maintenance to assure that the validity of experiments is not threatened and that high utilization of the facility is possible.

5.3.4 Training Equipment

5.3.4.1 GENERAL

The training equipments required for the APOLLO Program are based upon consideration of the training requirements, engineering feasibility, and cost. Existing devices will be utilized extensively to avoid costly duplication of equipments which are adequate for the APOLLO requirements. These equipments will be used primarily for the experience of, and adaptation to, the stresses of the space environment. The exact degree to which these devices contribute to effective training will not be known until actual space flight experience is gained. Early Mercury and Dyna-Soar missions will undoubtedly alter space flight training techniques and equipment. This fact makes it highly desirable, if not imperative, to stress flexibility in future training equipment design.

In view of the lack of specific knowledge of what aspects of the space mission must be reproduced in the training situation, the tendency is toward overdesign of the training devices; that is, attempting to simulate everything so as to avoid the omission of seemingly insignificant items which may actually be important. Technical and cost considerations preclude the possibility of completely simulating the space mission. The ability to simulate long term weightlessness in a simulator, although desirable, appears to be technically impossible. Combining all of the environmental stresses such as accelerations, vibrations, noise, heat and atmosphere, in a simulator may be considered technically feasible but the cost and complexity of such a device appears to be prohibitive.

Consideration has been given to the development of an exospheric flight trainer for use in training for future space missions. Such a device could duplicate at least a portion of the space vehicle flight profile and subject the trainee to stresses and tasks similar to those encountered on the actual mission. This would not remove the requirement for ground-based training equipment but would permit the trainee to experience conditions not reproducible on the ground. As in aircraft flight trainers, it would not be possible to practice critical emergency procedures because of the risks involved. The development cycle of such a device would be long, its cost would be high and its training value uncertain.

The primary device recommended for use in coordinated crew training for the APOLLO mission is the Mission Simulator. This device will provide a virtually complete simulation capability through all of the phases of the operational mission. Significant by their absence will be the acceleration and deceleration forces during launch and re-entry. However, training on the critical tasks during the launch and re-entry phases will be stressed in the utilization of the closed-loop centrifuge. The Mission Simulator will provide a complete dynamic simulation of all the displays and controls within the re-entry and mission modules. These modules may be used together or separately. The computation portion of the Mission Simulator will be sufficiently flexible to permit simple reprogramming to accommodate changes in the vehicle design. Visual systems will be provided to simulate the external visual field of the crew. If the selected vehicle configuration permits approach and landing profiles similar to conventional high-performance aircraft, a motion system will be provided to simulate those significant cues utilized by a pilot during this mission phase. In the case of a semi-ballistic re-entry vehicle the inclusion of a complete motion system is not required, however, limited motions simulating vibration and buffet will be provided. An instructor's station shall be included containing the controls and displays necessary to conduct the training session, insert malfunctions, and enable the monitoring and evaluation of the trainee's performance.

As mentioned previously, a human centrifuge, operating closed-loop, will be utilized for training in the critical tasks which must be performed under high accelerations. Although it would be desirable to permit the three crewmembers to train simultaneously during this acceleration environment, the required payload would be beyond the capabilities of existing centrifuges. Present information indicates that the vehicle commander will be required to perform most actively during launch and re-entry. Therefore, his station should be simulated to the fullest extent practicable in the centrifuge gondola. The other crewmembers should also receive training on the centrifuge with at least the mission-phase-relevant portions of their stations simulated in addition to necessary crosstraining. Open-loop operation of the centrifuge will be utilized as a conditioning device to increase the acceleration tolerance limits of the personnel and permit them to practice tasks such as breathing, speaking and other motor acts.

Although the Mission Simulator could be used for part-task and procedural training, scheduling and cost considerations will dictate the use of separate trainers for these functions. This will not only relieve the trainee load on the Mission Simulator but it will permit training to be conducted at locations other than the simulation laboratory, for example, in a classroom.

Additionally, these devices will be used extensively in lectures and demonstrations in such areas as the theory and operation of APOLLO subsystems and the detection and correction of subsystem malfunctions. This would include practice in performing preventive and non-scheduled maintenance.

5.3.4.2 MISSION SIMULATOR

The Mission Simulator required for the APOLLO training program has been derived through the consideration of many factors, of which some of more importance are the following:

1) Fidelity of simulation.
2) Flexibility to accommodate changes in vehicle design.
3) Sufficient accuracy of vehicle motion computation throughout all mission phases.
4) Determination of computation techniques requiring least equipment to implement.
5) Engineering feasibility not requiring technological breakthroughs.

For a glide vehicle, the landing phase would be performed in a manner which is an extension of aircraft dynamics. Human control of this phase will probably be an exacting task and very good simulation will be required for adequate training, and as an aid to final cockpit design. The control task and the required simulation will be similar, although somewhat simpler for the controlled re-entry of a semi-ballistic vehicle.

Considerable experience has now been gained in the in-flight simulation of the handling characteristics of advanced aircraft and space craft particularly during landing maneuvers. The development of the variable stability aircraft has largely been responsible. Such aircraft contain a programmer which automatically varies the vehicle characteristics and display parameters, thus simulating the flight characteristics of a proposed vehicle. The X-15 development program has made use of a three-axis variable stability T-33 aircraft. A one-axis variable stability F-94 and a one-axis (longitudinal) variable stability B-29 have been studied. The T-33 has been used by X-15 pilots to study simulated re-entries with various moments and inoperative dampers. Re-entry accelerations are maintained by programmed turns at constant altitude. Such aircraft may also be used as fixed base flight simulators. In addition to landing and re-entry problems such aircraft have or will serve as a test bed for such studies as the comparison of various kinds of side stick controllers, transfer functions for eye-tracking motions and pilot control saturation and its relation to pilot opinion of control ease. Such simulations will be used for Dyna-Soar and would be applicable to a glide vehicle APOLLO.

Although the formulation of the flight equations for the simulation of the APOLLO vehicle will not be detailed in this document, some of the pertinent aspects of the equations will be discussed in order to support the equipment concepts which are described later. The problem is to derive a set of equations, or a number of sets, that provide simulation of sufficient accuracy throughout the entire mission at the minimum equipment cost. Fortunately, some sets are clearly less advantageous than others and need not be seriously considered. Other sets (specifically, for the solution of the force equations) are not so easily evaluated. The equations for the APOLLO Mission Simulator were derived using the most promising axes systems with the final selection based upon the accuracy and cost of computation.

The Euler angle technique for defining relative orientation between axis systems is by far the simplest, requires the least equipment, and is the technique normally employed in aircraft flight simulators. However, a space vehicle operating out of the effective atmosphere can perform maneuvers which have no counterpart in aerodynamic flight. Specifically, a space vehicle is an all-attitude vehicle which can pitch, roll, and yaw through 360 degrees about all axes without influencing the velocity vector. The Euler angle technique has the one great disadvantage of angle ambiguity, or "gimbal lock", when $\Theta = 90$ degrees. Also, inaccurate projections result when Θ

is near 90 degrees due to computer scaling problems. The Euler angles may be defined such that "gimbal lock" does not occur when $\Theta = 90$ degrees, but the redefined system loses definition at some other angular orientation. The direction cosine and quaternion techniques eliminate these problems. Recent studies by Link Division of General Precision, Incorporated, show that a substantial saving in equipment can be effected by first computing quaternions from the vehicle's angular rates and then computing direction cosines from the quaternions rather than computing the direction cosines directly from the vehicle's angular rates.

In order to better define the computational requirements of the mission simulator, the expected form of the aerodynamic coefficient equations were investigated. The form of the coefficient equations for a glide-type re-entry vehicle was based on a review of preliminary Dyna-Soar aerodynamic derivative data. Because the vehicle configuration may exhibit appreciable elasticity, the equations of motion of the elastic modes were investigated and the effects of elasticity were added to the rigid body equations.

In addition to the implementation of the flight equations, subsystem simulation will be required in the Mission Simulator. These subsystems of the APOLLO vehicle included (1) instrumentation and communications, (2) navigation and guidance, (3) flight control, (4) electrical power, (5) crew services, (6) landing and recovery. In order to determine the performance characteristics of the equipment required to achieve a satisfactory simulation of the various vehicle subsystems, it was felt that unified design criteria should be generated and applied in turn to each of the vehicle subsystems and their elements. In this manner, the performance of each simulated element could be specified where complete data were available and reasonable estimates could be made in those areas where detailed information was lacking. The advantage of this approach is that it results in a unified set of subsystem simulation requirements with a corresponding reduction in the types of computing equipment and an increase in the flexibility of the system. Any vehicle subsystem can be specified by considering the two basic areas of input-output data and operating characteristics. The input-output data can then be classified into type, form, amount, and range of variables. The operating characteristics of prime interest are speed, accuracy, and resolution.

Some of the vehicle subsystems are required to generate or display information which is directly related to the vehicle's flight characteristics or its position in space. Some examples of this group include flight instrumentation, navigation displays, etc. The computational requirements for simulating this group of subsystems are essentially accommodated by the flight computations. Data conversion to the proper display format and malfunction insertion capability must, of course, be provided. The second category of vehicle subsystems include those which, although they may be indirectly dependent upon flight characteristics and conditions, essentially are characterized by their independent operation. Their simulation requirements, at best, are only partially accommodated by the flight computation. Examples of this type of subsystem include rockets, automatic flight control system, surface temperature instrumentation, electrical power system, etc.

Some of the principal design factors which must be considered in the determination of the best simulation technique are the rigor or fidelity of simulation, insertion of malfunctions, flexibility to permit adaptation of design changes, compatibility with the other portions of the simulator, and cost-complexity evaluations. The rigor of simulation and insertion of malfunctions are essentially defined by the human factor requirements. The rigor of simulation of a specific subsystem is determined primarily by the accuracy and resolution required in the outputs of the system. If malfunctions are not considered in the simulation of a subsystem, the system can be defined by a transfer function describing the operation of the system by relating the dependencies of the output variables upon the input variables. If simulated malfunctions are required, these malfunctions can be considered as additional inputs, or as changing the mode of operation of the basic system. In either event, the systems usually become considerably more complicated, sometimes sufficiently so as to change the basic approach to mechanization. In general, there should be a correlation between a simulated subsystem's relative importance, number of functions performed, accuracy and complexity. An overly-simplified simulation can negate the usefulness of the simulator as a training device while an overly-complex simulation, besides being more expensive, requires an unduly large portion of the equipment and thus reduces overall capability, increases failure rate, maintenance and "down time", and is usually more difficult to modify.

In order to determine the approximate computing capacity required in the APOLLO Mission Simulator and the best method of performing the computation, several of the major subsystems were considered in some detail. Among these were the automatic flight control system, boosters, temperature of the vehicle surface, and the energy management problem. Skin temperature simulation requires the determination of the temperature at the positions where the sensors are located. A complete temperature simulation throughout the history of a mission would be extremely complex and would not justify the quantity of equipment involved. Certain approximations or assumptions must be made in order to solve the temperature computations. The greatest difficulty concerning aerodynamic heating is caused by the present inability to predict the transition from laminar to turbulent flow and to accurately determine the conditions at the outer edge of the boundary layer. The state of the art in the energy management area is still in the fairly early stages of development at this time. Considerable work is being performed in this field and the Bell Aerosystems report, Study of Energy Management Simulation Techniques, by John Ryken and Robert W. Austin (Secret), gives an account of equations to be solved and equipment necessary for the simulation of some energy management problems.

The most critical and important element of the APOLLO Mission Simulator will be the computer complex. It is because of this fact that the preceding discussion of some of the important facets of the computational problem have been included in this document. The two extreme approaches to this implementation are the all-digital complex (general and special purpose) and the all-analog computer. In between lies the "hybrid", or combination digital-analog approach.

Digital computation has the advantage of providing very precise static accuracy and the capability of holding this accuracy without drift problems, i.e., excellent repeatability. The accuracy of a digital computer, however, is not absolute, and does depend upon the sampling rate. The rate of change of variables, in the simulation of the APOLLO vehicle, may vary from a fraction of a cycle per hour to a fraction of a second per cycle. If a hypothetical digital machine is assumed in which the word length is sufficient to avoid "round-off" errors, and a computation rate of 20 samples per second the number of samples per cycle of any variable can be estimated. For example, variables with a period of one hour would permit a rate of 72,000 samples per cycle, while variables with a period of one second will permit only 20 samples per cycle. If frequencies increase beyond this point the computer implementation can become particularly difficult, depending upon the accuracy required for the particular variable. Studies have shown that the accuracy increases almost linearly with sampling rate.

It is clear that there are digital computers available today that could handle these problems and produce reasonably satisfactory results. However, the cost per "unit of computation" becomes excessive. If economic considerations were not important, it would be feasible to implement the Mission Simulator with an all-digital computer complex. Returning to the variable with a frequency of one cycle per hour, the sampling rate of even a modest digital computer can provide excellent accuracy and drift free operation. Another major advantage of the digital computer is its relative ease of re-programming without any change of hardware.

The analog computer has the advantage of rapid computation and is capable of accommodating the real-time computation requirements. It also lends itself well to integration and multiplication, two of the most frequently repeated operations in the solution of flight equations. Further, the maximum size of the computer does not have to be predetermined. Analog equipment lends itself to the addition of functions and systems, as required, without the necessity of purchasing large blocks of equipment for which there is no clearly established need. However, the analog computer cannot achieve the accuracies obtainable in digital computation and, in addition, exhibits a tendency to drift over extended periods of time. Recent improvements in electronic multipliers, low-drift amplifiers, and high-precision components point to substantially higher accuracies. Programming an analog computer is normally more difficult, or at least more time-consuming, than programming a digital machine.

The logical intent of a hybrid computer approach is to combine the relative advantages of both digital and analog techniques, and to evolve an optimum design configuration. The ideal situation is to utilize each type of computer where it functions best. A vigorous application of this principle, however, does not necessarily yield an optimum design because of programming considerations and the complexity and expense of too-frequent analog-to-digital and digital-to-analog conversions. A hybrid computer may be composed of separate digital and analog pieces of equipment coupled together by appropriate converters, or it may be one piece of equipment inherently combining both digital and analog equipments and obtaining the conversion from one form to the other as part of the specific computation. Whatever configuration it may take, the hybrid computer can be a very powerful computational tool, supplying speed where speed is required and accuracy where accuracy is required.

Some of the inputs and outputs of the combined system, such as indicators and controls, will be analog devices which can be connected directly to an analog computer. The interconnection of analog and digital computers in a single, real-time simulation of the APOLLO vehicle permits those operations requiring digital techniques and those requiring analog techniques to be handled at the same time. For a real-time simulation, the digital computer must be fast enough to complete all of the required computations for a particular sampling interval within one interval, including all of the necessary input-output operations. If all of the operations of this complex simulation were to be performed in a digital computer the large number of required computations per sample combined with the required high sampling rate means that a present-day digital computer would encounter difficulty in maintaining the real-time scale. Accordingly, it is advantageous to relieve the digital computer of some of its computing load by performing some computations in the analog machine, particularly those involving high frequencies and moderate accuracies.

Based upon the above mentioned considerations, it is recommended that the computation portion of the APOLLO Mission Simulator be implemented by digital and analog (hybrid) techniques. This will give the best overall performance and greatest versatility per unit of cost. The primary advantages of this concept are high accuracy where accuracy is required; high speed where speed is required, and reasonable flexibility without undue size and complexity. It is believed that flexibility is extremely important in order to insure against obsolescence of the simulator and to permit changes in programming to conform to design changes in the APOLLO vehicle. Figure II-5-5 is a functional block diagram of the mission simulator.

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It is clear that there are digital computers available today that could handle these problems and produce reasonably satisfactory results. However, the cost per "unit of computation" becomes excessive. If economic considerations were not important, it would be feasible to implement the Mission Simulator with an all-digital computer complex. Returning to the variable with a frequency of one cycle per hour, the sampling rate of even a modest digital computer can provide excellent accuracy and drift free operation. Another major advantage of the digital computer is its relative ease of re-programming without any change of hardware.

The analog computer has the advantage of rapid computation and is capable of accommodating the real-time computation requirements. It also lends itself well to integration and multiplication, two of the most frequently repeated operations in the solution of flight equations. Further, the maximum size of the computer does not have to be predetermined. Analog equipment lends itself to the addition of functions and systems, as required, without the necessity of purchasing large blocks of equipment for which there is no clearly established need. However, the analog computer cannot achieve the accuracies obtainable in digital computation and, in addition, exhibits a tendency to drift over extended periods of time. Recent improvements in electronic multipliers, low-drift amplifiers, and high-precision components point to substantially higher accuracies. Programming an analog computer is normally more difficult, or at least more time-consuming, than programming a digital machine.

The logical intent of a hybrid computer approach is to combine the relative advantages of both digital and analog techniques, and to evolve an optimum design configuration. The ideal situation is to utilize each type of computer where it functions best. A vigorous application of this principle, however, does not necessarily yield an optimum design because of programming considerations and the complexity and expense of too-frequent analog-to-digital and digital-to-analog conversions. A hybrid computer may be composed of separate digital and analog pieces of equipment coupled together by appropriate converters, or it may be one piece of equipment inherently combining both digital and analog equipments and obtaining the conversion from one form to the other as part of the specific computation. Whatever configuration it may take, the hybrid computer can be a very powerful computational tool, supplying speed where speed is required and accuracy where accuracy is required.

Some of the inputs and outputs of the combined system, such as indicators and controls, will be analog devices which can be connected directly to an analog computer. The interconnection of analog and digital computers in a single, real-time simulation of the APOLLO vehicle permits those operations requiring digital techniques and those requiring analog techniques to be handled at the same time. For a real-time simulation, the digital computer must be fast enough to complete all of the required computations for a particular sampling interval within one interval, including all of the necessary input-output operations. If all of the operations of this complex simulation were to be performed in a digital computer the large number of required computations per sample combined with the required high sampling rate means that a present-day digital computer would encounter difficulty in maintaining the real-time scale. Accordingly, it is advantageous to relieve the digital computer of some of its computing load by performing some computations in the analog machine, particularly those involving high frequencies and moderate accuracies.

Based upon the above mentioned considerations, it is recommended that the computation portion of the APOLLO Mission Simulator be implemented by digital and analog (hybrid) techniques. This will give the best overall performance and greatest versatility per unit of cost. The primary advantages of this concept are high accuracy where accuracy is required; high speed where speed is required, and reasonable flexibility without undue size and complexity. It is believed that flexibility is extremely important in order to insure against obsolescence of the simulator and to permit changes in programming to conform to design changes in the APOLLO vehicle. Figure II-5-5 is a functional block diagram of the mission simulator.

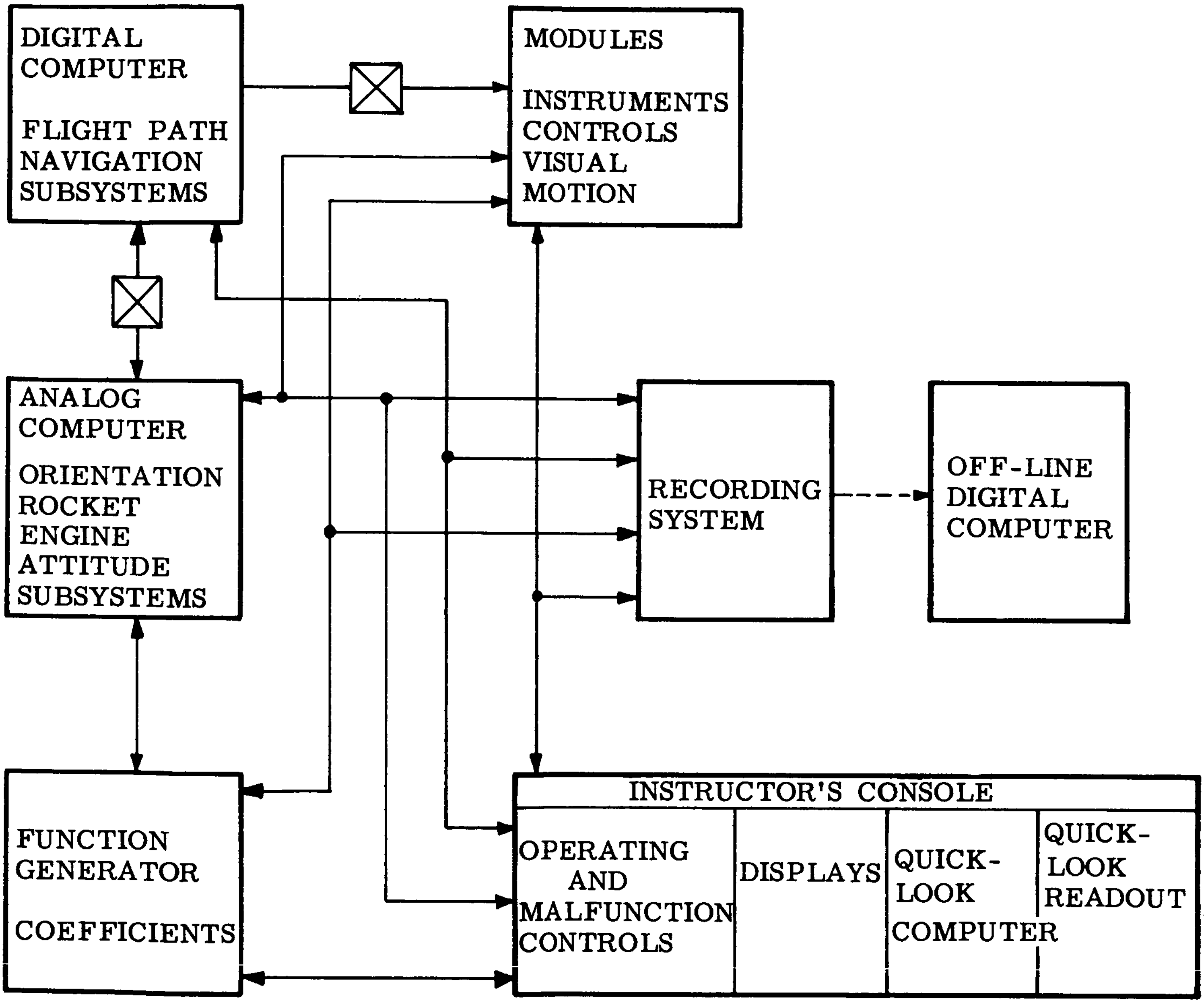

Figure II-5-5. Functional block diagram of mission simulator

The digital computer will be used to solve the aerodynamic force equations and to generate the simulated APOLLO vehicle location with respect to the earth (during earth-centered navigation) or with respect to the Moon (during Moon-centered navigation). In general the equations solved by the digital computer will have one or more of the following characteristics or requirements: 1) precise long-time integration of variables (flight trajectory and orbit); 2) determination of the difference between two large, nearly-equal variables (e.g., thrust and drag); and 3) variables having exceedingly wide ranges (such as altitude).

The aerodynamic and propulsion forces are inputs to the digital computer from the analog computer after conversion into digital form. The gravity forces are computed in the digital computer, taking into account the oblateness of the earth. The three coordinates of the sum of all forces are calculated, and acceleration, velocity, and the coordinates of the center of gravity of the vehicle are obtained as solutions. The trajectory and orbit of the vehicle are determined by the force equations; applying digital techniques for this section of the simulator makes it possible to achieve the high accuracy that is required. It is recognized that some of the inputs are derived in the analog computer and can have only the accuracy inherent in this computer. When the vehicle is outside of the effective atmosphere, the aerodynamic forces become negligible and only the gravity forces acting on the vehicle are significant unless thrust is applied to change the orbit or trajectory. The digital portion of the simulator computer will make use of the results of the navigation and guidance simulation program described in Chapter II of Volume III. Figure II-5-6 is a block diagram indicating the signal flow of the flight system equations. The signals shown are for a glide re-entry vehicle during earth orbit with the thrust and drag term subscripts of (f) and (m) denoting the axis system in which the force and moment equations, respectively, are solved.

The analog computer will solve the vehicle orientation equations and the many subsystem equations which are adapted to pure analog computation because of the nature of the equations or the form of the output information. This portion of the flight equations does not dictate the accuracy requirements as those equations recommended for digital computation. The frequency requirements, however, are much higher in this area—in the order of 500 degrees per second for some variables. It should also be

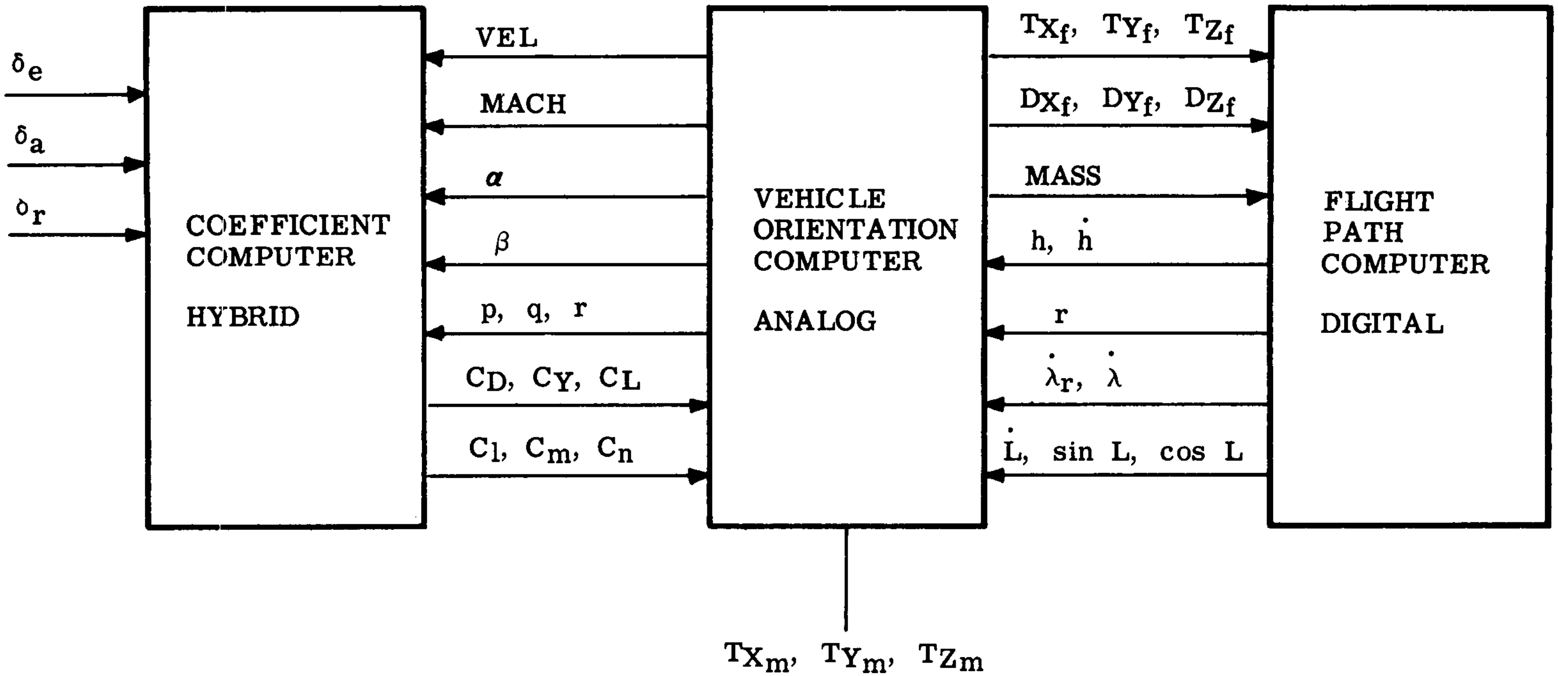

Figure II-5-6. Flight system signal flow diagram noted that there are a large number of multiplication of variables in these computations. In the past, these multiplications have normally been handled by servos. However, servos suffer from non-linear effects due to backlash, static friction, and wire-to-wire resolution, and all of these effects tend to reduce or destroy the response to small perturbations. For this reason, high-accuracy electronic multipliers will be used in place of servos. It should also be mentioned that the orientation portion of the flight system is fixed and not a function of the vehicle characteristics; therefore, this portion of the flight computer need never be reprogrammed.

A hybrid function generator will solve the flight coefficient equations and the vehicle bending-mode equations. In the event that future vehicle configurations include a jet engine for approach and landing, these equations would also be solved in this function generator. Each of these three areas contain a large number of non-linear functions. The frequencies may be quite high (on the order of 20 cps) in the flight coefficients and the bending-mode equations. The hybrid function generator must be capable of handling such frequencies as may be encountered. The characteristics of the vehicle are determined by the equations handled by this device and, therefore, it must be capable of being reprogrammed with reasonable ease.

The simulator re-entry and mission modules will be constructed such that the interior configuration and dimensions are identical to those of the APOLLO vehicle. All controls, displays, and interior furnishings will be identical in appearance and operation to those in the actual vehicle. There is no need, nor is it desirable, to duplicate the external dimensions of the vehicle. Instead, the exterior of the simulator should be of a durable material, fabricated in such a manner as to facilitate access to the interior for the purposes of changing or modifying instruments, controls, etc. Flexibility must be stressed in the design of the modules in order to permit changes in the simulator to conform to design changes in the APOLLO vehicle.

The simulator re-entry and mission modules may be used together to provide co-ordinated crew training throughout all mission phases. In addition, it will be possible to utilize the modules separately for training crew members on tasks which are relevant to certain mission phases. For example, one crew could be practicing on re-entry problems in the re-entry module while other crew members are receiving training on mission module subsystem operation.

It would appear desirable to maintain the same relative orientation between the two modules as in the actual vehicle. The only task, however, that seems to be affected by this orientation is that of passage through the tunnel connecting the two modules. Since all crew members will be in the re-entry module during periods when accelerations are significant, practice at the one G environment of the simulator would be of little value. Of much more significance is the fact that the "floor" of the mission module may not be oriented the same as the "floor" of the re-entry module. It is recognized that this is not a requirement for the "weightless" environment of space but it obviously is a requirement for a ground-based simulator. Therefore, the two modules will be positioned such that their "floors" are parallel to the floor of the simulation laboratory.

The simulation of the external visual field of the crew members is one of the more difficult areas in which to achieve realistic results. It appears that the use of television techniques will result in the most flexible system to accommodate the various displays which may be required. The earth display would be provided by means of a spherical model showing the significant topographic details. The model would be rotated according to the vehicle position and a light source would serve to indicate the areas of day and night. A prism scanner could be interposed between the earth model and the television camera to facilitate rapid apparent motion of the earth due to vehicle tumbling. The camera would be equipped with a zoom lens to simulate changes in apparent size due to altitude variations. A similar model of the Moon would be used during circumlunar and lunar-orbital missions. Of course, the portion of the Moon which is not visible from the earth could not be represented at this time, however, instrumented satellite shots should provide considerable information prior to the first manned mission. It should be mentioned that it does not appear feasible to represent the earth or the Moon to the level of detail which could be seen, for example, through a telescope aboard the vehicle being used for scientific observation. Even if the earth (or Moon) were known to that level of detail, the amount of information which would have to be placed on the model is enormous.

Another display which should be provided is that required for navigation training. This unit would consist of a hemispherical translucent screen on which images of stars, planets. sun and Moon are projected in a manner similar to that of a conventional planetarium. The two types of motion which are required is the rotation of the entire star framework due to attitude changes of the vehicle and the motion of the members of the solar system within the framework of the fixed stars in response to the position of the vehicle within the solar system.

If the selected vehicle configuration were of the glide re-entry type with landing characteristics similar to that of an aircraft, an approach and landing display is required. Of the many landing displays which have been developed for use with flight simulators, the type which mounts a terrain model on a moving belt and scans it with a television camera is recommended for use with the APOLLO Mission Simulator.

If the re-entry vehicle is of the type described above, a motion system should be provided. This system would provide motion only to the re-entry module. The most advanced motion system presently utilized with high performance aircraft flight simulators would appear satisfactory for this application. This system will provide 20 degrees of pitch rotation, 18 degrees of roll rotation, 24 inches of vertical translation, and vertical accelerations of 0.8 g's above or below the normal 1G. Further, the vertical translation can occur without any coincident pitch angular rotation.

An instructor's console will be provided containing all of the controls necessary to operate the mission simulator equipment and to insert malfunctions into the simulated vehicle systems. The console would also contain indicators which would repeat much of the information which is displayed to the crew. The indicators would not necessarily be of the same form since it is often desirable to utilize less complex, and less expensive, instruments at the instructor's console than are required in the vehicle simulator. Recorders and plotters would be provided to enable the instructor to monitor the overall progress of the flight. These might be in the form of X-Y plotters, approach recorders, and flight path recorders. To display the vehicle's position during earth-centered navigation, it appears feasible to design a map projection device that contains two hollow transparent segments of spheres on which the map of the earth has been printed. Each segment would be slightly more than a hemisphere and, thus, contain a map of somewhat more than one-half of the earth's surface. A point light source would be located at the center of each sphere segment to project the selected area of the earth onto the display surface. The sphere segments would be rotated about their centers in response to the vehicle's latitude and longitude which is available in the digital portion of the simulator computer. Vehicle position would always be shown at the center of the display surface (a vehicle symbol could be superimposed or etched upon the surface) resulting in minimum map distortion at the point of interest, i.e., the vehicle position. This device would display, essentially, a gnomonic map projection which is characterized by minimum distortion at its center. A true gnomonic projection has zero distortion at its center, but the necessarily finite size of the light source will result in some distortion even at the center.

In order to assist the instructor in the scoring and evaluation of the crew member's performance, a small general-purpose digital computer should be utilized. This unit would not be used to process simple readouts but for the reduction of data to obtain "quick-look" scores. Some readout in hard copy, e.g., Flexowriter, would be provided to relieve the instructor of the necessity of transcribing meter readings, determining the time to perform tasks, etc. It is also recommended that magnetic tape transports be used to record all of the important data throughout the mission for later off-line data processing. It is not believed necessary to procure the off-line computer as part of the simulation facility. Any large general-purpose computer that would be available for short periods of time could perform this function.

5.3.4.3 VEHICLE COMMANDER'S STATION

It appears that the most critical tasks during high accelerations will be performed by the vehicle commander. Therefore, his station should be simulated to the fullest extent practicable in the gondola of a human centrifuge. All of the controls and displays utilized in the launch and re-entry phases of the APOLLO mission should be operable. The centrifuge would be operated in a closed-loop consisting of the vehicle commander's controls and displays, the centrifuge, and a computer. This crew member will be part of the system to the extent that he will interact with the computer and its aerodynamic equations so that his performance will affect the displays, controls, and the acceleration profiles. The acceleration profile of the boost phase can be approximated quite closely on existing equipment, although it does not appear to be possible to exactly duplicate the rapid drop-off of acceleration at staging.

If other members of the crew are required to perform important functions during the launch and re-entry phases, their stations should be mocked-up to the extent necessary to permit them to practice these tasks while undergoing the accelerations experienced in these mission phases. Due to the payload limitations of existing human centrifuges it does not appear possible to train the three crew members simultaneously. During open-loop operation of the centrifuge, either for task training or conditioning, the centrifuge will be driven by programmed signals. These signals may best be derived by recording the centrifuge operation when it is under closed-loop control by the vehicle commander. In this manner "realistic" re-entry profiles can be experienced by all members of the crew.

5.3.4.4 PART-TASK AND PROCEDURES TRAINERS

Part-task and procedures trainers will be required for the APOLLO training program primarily for use in training on the theory and operation of vehicle subsystems and the detection and correction of malfunctions. The correction of malfunctions would include training in equipment maintenance. The number and types of these trainers cannot be specified at this time since they are dependent upon the detailed design and operation of the specific vehicle subsystems. This type of information would become available during the preliminary design and developmental engineering phases of the APOLLO program. Some of the general characteristics of these types of trainers will be described briefly.

The devices used for training in the theory and operation of vehicle subsystems can range from simple static models to relatively complex simulations of the operational systems. The characteristics of procedural or part-task trainers are: 1) a mock-up of part or all of the trainee's work station, 2) only the controls and displays necessary to train for a single task, or closely related tasks, are operational, 3) the dynamic response of controls and displays are only as accurate as is required to accomplish the desired training and 4) the actual equipment operation is simulated only to the extent required for proper control-display interaction and the insertion of certain malfunctions.

Other devices will be required to provide training in maintenance tasks. A particular maintenance trainer will depend greatly upon the type of maintenance that can be performed. If the maintenance process is simply to remove and replace modules, then only the parts of the equipment that are accessible for diagnosis must be provided. The replaceable modules can be simple mock-ups. If troubleshooting and the replacement of failed components can be accomplished below the "black box" level, a more complete representation may be required. It may even be best to utilize part, or all, of the operational equipment. However, the use of a maintenance trainer in lieu of operational equipment provides many advantages such as: 1) a maintenance trainer can be simply designed to be more reliable than the operating equipment and, thus, reduce troublesome maintenance and downtime; 2) all malfunctions can be controlled; 3) malfunctions have no damaging effects on associated equipments; 4) trainees can work in safety, e.g., no exposure to high voltage; and 5) training can be active, and, hence, more meaningful. Also, in a development program such as APOLLO, difficulties could be anticipated in obtaining sufficient operational equipment to conduct maintenance training, particularly during the earlier stages of the program.

5.3.4.5 ADAPTATION DEVICES

Certain devices will be required in order to acquaint the crew members with the stresses of the space environment and to provide some adaptation to those stresses. The extent to which these devices contribute to effective training will be better known after initial space missions, such as Mercury, and it is recommended that existing devices be considered for use in the APOLLO program unless future data shows the need for additional equipments. Such devices as the AMAL centrifuge and the Lewis three-gimbaled tumbling device would appear to be definitely useful in the training program. Some of the environmental stresses could be simulated in the mission simulator and permit the crew to practice complex tasks while experiencing some of the combined stresses. It appears feasible to provide vibration, buffet, noise, and temperature effects in the mission simulator and, thus, eliminate the necessity of using other devices for these aspects of the environment.

5.3.4.6 OTHER TRAINING AIDS

Other training aids will be utilized in the APOLLO training program particularly in classroom training and individual study. These would include slide and motion picture projector, teaching machines, tape recorders, manuals, cut-away assemblies, mock-ups, and models. The types, numbers, characteristics, etc. of these training aids would be determined as the detailed training program is developed.

5.4 SIMULATION AS A DESIGN TOOL

The APOLLO Mission Simulator and part task simulators can furnish the data needed for solving many troublesome but important problems in the human factors area. Data can be collected in the simulators under controlled, yet realistic conditions. These data, relating crew and vehicle performance to variables in each of the problem areas listed below, make informed decisions possible, not only concerning the relative merits of alternate solutions to a given problem, but, also, whether a given problem approach will meet the requirements of the situation.

5.4.1 Allocation of Functions between Equipment and Crew

While aircraft are the closest relatives in our experience to manned spacecraft, there are enough differences to require reevaluation of all the man-machine control systems. Which, if any, of the functions that are performed by man in a conventional aircraft can not be executed rapidly enough or accurately enough in the APOLLO vehicle? Which functions previously automatic may now be made safely manual? An answer to these and other questions about function allocation is best obtained in a well-controlled experimental situation, rather than by resorting to extrapolation from experience with aircraft or basing decisions on unsupported opinions.

5.4.2 Layout of Capsule

Whatever functions are allocated to the crew, it will be important that they be performed well. This means that the information requirements must be properly met and the entire interior layout of the capsule must be designed to facilitate performance by the crew. Furthermore, this will require that crew stations be comfortable and accessible, that instruments and other displays are within the range of vision and easily used, that controls be within easy reach and simple to operate, and that these components be arranged according to frequency of use, sequence of use, and functional relatedness.

5.4.3 Layout of Other Portions of the Vehicle System

Although most of the human factors effort in both industry and the military, has centered about the control station, and simulators have been generally restricted to cockpit sections, other parts of the vehicle system, if simulated, could be improved in layout in much the same way as the cockpit.

For example, although engineering imperatives govern much of the placement of airborne equipments, small differences in layout can yield large differences in the ease with which these equipments can be maintained. An integrated and comprehensive simulation facility should have provision for testing the maintenance as well as the operation of the aircraft.

Another example is the ground support equipment. Close coordination of the ground support equipments with those of the vehicle are required for safety. Interface problems, some not easily predictable, can be handled expeditiously with comprehensive simulation facility.

5.4.4 Evaluation of New Instruments and Controls

New displays, new instrumentation, and new controls will be required to facilitate decision making at a much more rapid pace. "Quickened" displays and "predictor instruments" may be needed. Readouts from digital computation can be expected to replace many analog type displays. The logical place to prove out these new equipments and concepts is in the simulator where the evaluation can be performed safely, inexpensively, and, because of the precise experimental control, soundly.

5.4.5 Specification of Work Environment

The work environment of the crew must be carefully controlled to maintain the personnel at peak working effectiveness. If no light is admitted to the vehicle from the outside, the problem of interior illumination becomes very different from that of an airplane. Problems of heat, vibration, buffetting, and pressure altitude also require attention.

5.4.6 Personnel Selection

It is probable that different capabilities will be required for personnel manning APOLLO from those needed for conventional aircraft or even Mercury. The use of the simulator will enable selection standards and procedures to be set up in advance of the initial training of operational crews.

5.4.7 Development of Operational Procedures

Many procedures adequate for aircraft are too time-consuming or otherwise inappropriate for APOLLO. Procedures, essentially new and different from those in airplane control must be worked out in great detail before an APOLLO mission can be flown; they cannot be improvised.

5.4.8 Allocation of Tasks Among Crew Members

In much the same way that operational procedures are developed and evaluated in the simulator, the division of labor, or allocation of tasks, among the crew members can be made. Again it is important that each task should be assigned to one crewman, so that no task is left undone, and none is executed poorly because two men attempted to accomplish it simultaneously, but only interfere with each other. Judicious assignment of tasks will also prevent overloading any one crewman.

In summary, it is emphasized that appropriate use of such a simulation facility will make major contributions to the design and development of an effective operational vehicle and man-machine system.

5.4.9 Additional Human Factors Research and Development Requirements in the Training Area

In addition to the research and development described above, that can easily be conducted using the APOLLO Mission Simulator, a number of other research and development requirement exist in the training area.

5.4.9.1 TRAINING CONCEPTS

New techniques are required for more closely specifying training content, procedure, and methods to assure adequate skill acquisition at minimal cost. New principles and training concepts may need to be developed. With current techniques, a large degree of over-training is usually required to assure meeting of standards.

5.4.9.2 SELECTION OF TRAINING MEDIA

Methods are required to determine which training media (methods and devices) are most appropriate for a given subject matter area and for given objectives. Currently, intuition and tradition are heavily relied on in setting up training media for training programs.

5.4.9.3 PROFICIENCY MEASUREMENT

Better evaluation devices are required both for training feedback and administrative evaluation. Current measurement techniques are often too low in reliability, validity, practicality, and acceptability.

5.4.9.4 TRAINING WHILE IN FLIGHT

During a two-week mission, not only may the astronaut lose skills and knowledge through forgetting, but additional material may have to be learned. Methods are therefore needed not only to maintain knowledge and proficiency during a mission, but to train the astronaut in new areas. Application of automated instruction appears to be the direction a solution to these problems will take. Programmed instruction may also be desirable to provide useful activity for the astronauts to offset boredom during a long mission.

5.4.9.5 SPECIFICATION OF FIDELITY OF SIMULATION

Both for part-task trainers and mission simulators, no explicit procedures exist for determining, before the fact, what the shape of the transfer of training versus fidelity of simulation curve will be. Hence, a conservative approach often results in the specification of higher fidelity of simulation than is necessary, at increased cost. Sometimes, when budgetary limitations are severe, fidelity is sacrificed to keep costs down. With better knowledge of the relationship between fidelity and training value, both of these extreme positions could be avoided, and adequate training can be assured at minimal cost.

6.0 Psychological Considerations

6.1 INTRODUCTION

In addition to the factors previously considered in this volume, there are a number of additional parameters which influence human performance. These factors are perhaps more subtle in their influence, and have received less attention from Human Factors Scientists than matters such as selection, training, and control-display design. However, a growing body of knowledge indicates their importance. Recognition of these factors, and their application to vehicle hardware design and operation procedures will clearly contribute to crew performance capabilities.

In this section we have defined many of these problem areas, indicated possible solutions, implications for vehicle design, and have suggested research and development work which would contribute to the overall vehicle system design. The application of the information presented here will become increasingly important as consideration is given to operational design and the crew activities aboard the APOLLO vehicle.

6.2 THE EXPERIMENTAL ANALYSIS OF PERFORMANCE

Detailed analysis of performance decrement became more sophisticated during and after World War II with the studies of Bartlett (1942a, 1942b), Mackworth (1950), Broadbent (1953), Wilkinson (1958), and others at Cambridge, England. These investigators view the human operator as a kind of data processing system able to perceive, select, perform mental operations on, store, and retrieve data. Adverse environmental conditions like heat, noise, acceleration, low oxygen pressure could affect this system at any point. The problem for the investigator is to find out how and where in the system the breakdown is occurring. Solving this problem provides basic information about behavior as well as methods for minimizing performance decrement.

What is the general nature of impairment in performance? Recent studies (Bartlett, 1942; Broadbent, 1955; Rosvold et al. 1956; Hauty and Payne, 1956; Maag, 1957; Williams et al. 1959) indicate that for a large number of conditions (e.g., sleep-deprivation, brain injury, fatigue, hypoxia, high nitrogen pressure, boredom and certain tranquilizing drugs) impairment of performance takes the form of an increasing irregularity or unevenness. The operator cannot maintain performance at a constant high level. Instead, from time to time his performance falters or stops. These "lapses" increase in frequency and duration as the adverse conditions are prolonged or intensified.

The physiological correlates of these lapses probably vary from one damaging condition to another. In sleep deprivation, the major factor is the occurrence of brief periods of extreme drowsiness or sleep. Lapses account for much of the performance decrement observed during experiments on sleep-deprivation. After 60 to 80 hours of sleep loss, lapses are often associated with periods of aberrant perception and dreamy states.

Lapses during drowsy states are signalled by, and accompanied by, characteristic EEG changes, increasing finger pulse amplitude, slowed heart rate, slowed breathing, and lowered muscle tonus. Between lapses, the operator is able to perform at, or close to, his usual level.

When decrement appears in the form of intermittent lapses, what aspect of performance will be most sensitive to the adverse condition? Psychologists usually group tasks in terms of the common mental process they are thought to represent, e.g., spatial tasks, verbal reasoning, learning, etc. The lapse hypothesis generates a different kind of grouping. Following Broadbent (1953) we classify most tasks as self-paced or work-paced. In self-paced tasks like sending messages, taking navigational fixes, plotting data, or solving arithmetic problems, the operator controls four important variables: (a) the time when the signals or instructions appear, (b) the duration of the signals, (c) the speed at which he proceeds from one part of the problem to another, and (d) his response time. In work-paced tasks like radar observation, monitoring communications, or receiving orders, these four aspects of the task are controlled by the work-situation or by a machine. There are tasks like tracking or pilotry in which some portions are self-paced and others are work-paced. In tracking, for example, the time when the signal occurs is work-paced, but other variables are self-paced.

In a self-paced task, the response can be deferred until a lapse is over, or the operator can begin again. Since highly motivated subjects will strive for accuracy, performance decrement will appear as a change in speed or output, but not in accuracy. In a work-paced task where the signal is present for a limited time, and the effective response time is short, a lapse coinciding with the appearance of a signal will ordinarily produce an error of omission.

Table II-6-I presents a summary of vehicle related tasks which are differentially sensitive to adverse environment effects.

TABLE II-6-I. VEHICLE-RELATED TASKS SHOWING DIFFERENTIAL SENSITIVITY TO ADVERSE ENVIRONMENTAL EFFECTS

Area of Work	Sensitive Task	Insensitive Task
Computation	1. Work paced	1. Paced slowly, or self-paced, permitting 100% correct responses
	2. Multiple choice answer	2. Feedback for incorrect responses
	3. Visual display	
	4. Measure time	3. Measure accuracy
	5. Task duration exceeds 1/2 hour	4. Short work periods, 2-15 minute rests
Monitoring	1. No neutral signals	1. Dual modality
	2. Low response rate	2. High response rate
	3. No feedback	3. Feedback
	4. Long duration	4. Short duration, 2-15 minutes
	5. Operator immobilized	
	6. Measure errors of omission	5. Measure errors of commission
Sequential cognitive monitoring	1. Work paced	1. Self-paced, long response times permitted
	2. Each response dependent on prior responses	2. Feedback
	3. Long duration	3. Responses independent
Tracking	1. Discontinuous inputs	1. Slow regular inputs
	2. Near threshold signals	2. Strong signals
	3. Measure errors of commission	

6.3 FACTORS AFFECTING SUSTAINED PERFORMANCE

What task conditions promote or modify impairment? Are there changes in the performance situation which will minimize or maximize lapses? Several such procedural factors have been studied: task duration, rest pauses, monotony, warning signals, knowledge of results, information load, motor load, drugs, and physiological day-night cycle.

6.3.1 Task Duration

In general, lapses increase both as a function of the stress situation and of task duration. Drowsy, or bored, or toxic subjects can pull themselves together to work very well for a few seconds or minutes but show increasing impairment as the task increases in length. Brief tests, or changing task conditions have a recuperative effect, but Hauty (1959) found that with extremely prolonged performance, decrement was not completely dissipated even with a normal period of sleep. When the operator resumed work, he seemed completely rested, but as work continued, decrement occurred sooner and progressed at a greater rate than during the previous work period. Thus, the effects of operator fatigue are cumulative. (The effect of drugs on sustaining performance is discussed below.)

Hauty (1959) and Williams et al (1959) agree that the important factor in task duration is not simply the total work time. Tasks with considerable variety such as complex problem solving hold up surprisingly well with fatigue and sleep loss. The relevant factor is the duration of relatively constant stimulus-response condition; that is, monotony. When a subject who is highly motivated to achieve and maintain the highest proficiency is forced to confine his complete attention to a rather small perceptual field where required operations are highly repetitious, performance decrement will occur early and progress rapidly.

6.3.2 Warning Signals

During prolonged performance, it is helpful to warn the operator prior to each signal. Lubin (1957 unpublished) found that during 72 hours of sleep loss, geometric stimuli could be perceived and recognized with high accuracy, if the experimenter warned the subject before each presentation with the command "fixate." Hauty (1959) found with a complex visual monitoring task, auditory warning signals were more effective than visual, and that warning signals which identified the specific indicators requiring action were better than general warning signals. The gains in sustained performance were not transient, but lasted throughout prolonged work.

6.3.3 Knowledge of Results

In general, knowledge of results reduces the impairment due to stress, but the results have not been entirely consistent from one task to another. Mackworth (1950) in his fatigue studies, and Wilkinson (1957) in his studies of sleep loss, demonstrated strong effects of knowledge of results. In studies by Williams, et. al., of sleep loss, the effect was present but not strong. The source of this discrepancy may lie in the way in which knowledge of results is given. In one study, the subject saw his reaction-time after each trial and had to make his own judgement about what was satisfactory. Presumably the practice runs he got gave him some idea of his level but no definite standards were set for him. This procedure did not prevent sleep loss decrement. In Mackworth's task, the procedure was to say to the subject after each trial, "Yes, that's right," or "You missed that one." This procedure was very effective in reducing decrement.

Goodnow (Williams et. al., 1959, p. 10) showed that if the goal of the task (e.g., errorless performance) can be changed by the subject, sleep loss will produce a decline in the subject's standards of performance. Hauty found that with fatigue there is an increasing range of indifference. F. C. Bartlett (1942a) noted the same phenomenon. Early in the work period the operator permits only slight deviations from optimal performance but as work continues his standard of performance becomes lower and lower. This lowering of standards may occur with or without the operator's awareness. Presumably reminders of failure and acknowledgment of success will help to prevent shifting standards.

6.3.4 Information Load, Cognitive Load, and Motor Load

Three other important intrinsic aspects of the task are the information load (i. e., the uncertainty of the perceptual display in terms of bits per time period), the motor load (i. e., responses per time period), and the cognitive load (i. e., amount of information to be stored, or number of mental operations per time period). In general, results show that (within reasonable limits) increasing motor load tends to reduce impairment. Simple motor tasks like tapping as fast as possible are not affected by sleep loss. Metz (1960, personal communication) of Strasbourg, France, found that with sleep loss, heavy muscular exercise improved performance on a complex monitoring task. Hauty (1959) found, however, that with prolonged performance on a complex visual motor monitoring task, several components of motor skill are affected. There is a loss in timing of sequential motor acts. The fatigued operator loses his ability to program such a sequence. In a long sequence, he may execute the right response at the wrong time. Presumably, the longer the sequence or more complex the required motor integration, the more pronounced the performance decrement.

The effect of increased uncertainty in the stimulus display depends upon the particular stress being studied. Detection performance on complex vigilance tasks in which several dials, clocks or lights were monitored often has shown no decrement as a function of fatigue (time on watch) (Jerison and Wallace, 1957). The additional display complexity, however, usually means that a considerable number of errors occur for the unfatigued subject. With sleep loss, increasing the uncertainty of a simple perceptual display has generally led to a greater absolute performance decrement (Lubin et. al., 1960, unpublished).

What happens if the task is made completely redundant? For work-paced tasks, results have not been completely consistent, but it seems clear that for self-generated, completely learned motor sequences, there is little sensitivity to sleep loss. Counting, repeating the alphabet, or pressing a lever over and over can continue through deep periods of drowsiness, and even through brief epileptic seizures (A. Mirsky, 1960, personal communication). If, however, the subject is required to stop or change the automatic sequence, he has great difficulty doing so, at least during drowsy states.

Williams et. al, have studied the interaction of increased cognitive load with sleep deprivation. In general, when the number of mental operations, or the amount of information to be stored per unit of time is increased, the task shows greater decrement. The drowsy operator cannot process or store data as fast as the normal operator. It is believed that this result will be found with hypoxia, anxiety, and other stress conditions.

6.3.5 Drugs

The influence of drugs in counteracting the effects of fatigue and sleep loss has been truly impressive. Mackworth (1950) and Solandt and Partridge (1946) have demonstrated conclusively that the oral administration of 10 mg. of amphetamine sulphate (Benzedrine) one hour prior to watchstanding abolished performance decrement on a vigilance task. The drug maintained its effect for at least 8 hours. Hauty (1959) in studies of prolonged work (7 to 30 hours), showed that a standard 5 mg. dose of d-amphetamine (Dexedrine) would sustain initial levels of proficiency for 7 hours of prolonged work, and would restore proficiency to baseline when administered after 24 hours of work. There were no detectable side effects like irrationality of judgment, lowered threshold or physical distress. Tyler (1946) found that Benzedrine would sustain performance during 100 hours of sleep loss. Williams et. al., found that with a single subject administration of high doses of Ritalin (Benzilic Acid, Diethylaminoethyl Ester Hydrochloride) during 135 to 200 hours of sleep loss did not return performance to baseline levels. Performance did improve, however, to a level roughly comparable to 72 to 96 hours of sleep loss.

Drugs appear to push performance closer to the limits of man's ability but after a point they are ineffective. Thus, the organism appears to have a built-in defense against being stressed too severely.

Benzedrine toxicity with chronic high dosages results in delirium characterized by visual hallucinations, delusions of persecution, confabulation and disorientation.

Ritalin is not believed to have severe toxic side effects. It has been used successfully with very high chronic doses in the treatment of narcolepsy by workers at the Mayo Clinic, with no notable side effects.

6.3.6 Physiological Day-Night Cycle

The physiological day-night cycle in man is most overtly manifested by the alternate phases of sleep and wakefulness. Since human babies and undomesticated animals are polycyclic (i. e., several periods of sleep during 24 hours), Kleitman (1939) believes that the monocycle observed in adult humans is determined more by social factors than by internal controls or by changes in the physical environment.

Physiological manifestations of this 24 hour cycle in man are evidenced by all systems and functions which regulate or contribute to metabolism. The temporal course of these manifestations is perhaps best illustrated by body temperature. Temperature has the clearest correlation with performance although there are individual differences. Changes in temperature appear to lead changes in performance by one-half to one hour. Depending upon the given individual's daily schedule of activity, the plotting of hourly readings for a 24-hour period reveals a monophasic (sometimes diphasic) cycle with maximum temperatures occurring during the regular period of wakefulness and minimum during normal sleep hours. While several "types" of diurnal temperature curves have been described, the most common one occurring in about 85% of the population, seems to be a rise in the morning, a slight fall in the afternoon or evening, with the lowest point between midnight and dawn.

When highly motivated volunteers are required to perform an exacting task demanding vigilance and judgment for 24 consecutive hours (Hauty, 1959), the resulting performance curve looks very much like the temperature curve. The sharpest onset of decline, and the lowest point of decline are reached during normal sleeping periods, at the usual waking up time.

Although the diurnal pattern of performance is fairly constant from day to day in the same person, there are wide individual differences. Kleitman (1949) suggested that individuals with wide diurnal temperature swings (i. e., with a "strong" diurnal rhythm) would be superior performers during the high temperature phase.

This relation of temperature and performance appears to be confirmed when one examines the work of Williams, Brady, Kleitman, Hauty, and Broadbent. This relationship is fairly well marked for monitoring type tasks. However, it seems that highly motivated individuals can "beat" the temperature cycle, at least during acute sleep loss. There is, however, no evidence to support the hypothetical relation of the strength of the cycle being related to greater performance swings.

It appears that man is committed to this diurnal rhythm. It can be shifted, reversed, lengthened and shortened to some extent, but not eliminated. Attempts to establish body temperature rhythms of more or less than 24 hours led to success in some subjects and failure in others (Kleitman, 1939). The length of the artificial period was important. Adjustment was made to 21 and 28 hour periods (8 and 6 day week) but not to 48 to 12 hour periods. Shifting or inversion of the day-night cycle is relatively easy; such shifts requiring several days to a week to achieve. Inversions of the temperature curve have been observed in night factory workers and, of course, it is well known that fairly rapid adjustment is made to time zone changes after long flights. The time required for adaptation may be a function of chronological age; younger subjects adapt more rapidly.

Body temperature and performance correlate well. It is probable, however, that not all types of performance follow the curve of body temperature. Kleitman et. al. (1938) showed a strong effect of body temperature on reaction time. The effect was greater for complex than for simple reaction time. A wide variety of laboratory tasks (Kleitman, 1939) such as dealing playing cards, mirror drawing, copying texts, transcribing codes, multiplying numbers, and color naming showed a typical diurnal curve. Especially interesting for this discussion is Kleitman's (1938, p. 225) finding that the number of blocks (lapses) per minute in a self-paced vigilance task was correlated with body temperature. The number of blocks per minute reached its lowest point in the afternoon tests where temperature peaked.

Hauty (1960) found that with prolonged performance, some tasks showed more decrement than others during low temperature periods. Greatest deterioration in efficiency occurred on a visual vigilance task, and a radar reconnaissance task. In problem solving and discrimination tasks where critical events were signalled by lights, deterioration was less marked.

Tasks which are resistant to dips in the temperature cycle and in general to adverse environmental effects include:

1. Automatic response sequences like lever pressing and tapping.
2. Tasks of short duration like threshold measurements.
3. Tasks where warnings are given.

4. Short intelligence tests.

5. Tasks with a great deal of variety.

If long duration tasks must be performed, the following techniques can be used to maintain performance in the face of a number of adverse environment factors.

1. Drugs.

2. Distinct warning signals.

3. Dual mode signals.

4. Converting the tasks to self-paced ones.

5. Scheduling frequent "breaks."

As expected, self-ratings on sleepiness and fatigue are correlated with the diurnal temperature cycle (Murray <u>et. al</u>., 1958). Psychotic behavior associated with prolonged (201 hours) acute sleep-deprivation has been observed to follow a diurnal rhythm. The subject reached a peak in disorientation, delusions, dread, and hallucinations between midnight and dawn, but these symptoms disappeared by 10 to 12 A. M., reappearing around midnight. Sleep-deprivation has tended to increase the amplitude of the diurnal temperature cycle rather than flatten it out.

As the detailed definition of APOLLO tasks becomes available the nature of the operations can be reviewed and it will be possible to design equipment and operating procedures so as to reduce or prevent task decrement. Various techniques are suggested to minimize the effects of adverse environmental factors. Thus, lapses in performance can be minimized by exercise, frequent change of jobs, immediate feedback on performance, warning when dials are reaching critical levels, dual-mode monitoring (simultaneous visual and auditory input), frequent rest periods, more than one observer, increasing the signal to noise rates, and analeptic drugs like dexedrine (McGrath <u>et. al.</u>, 1959). Feedback from ground stations would be desirable.

If accuracy is essential but speed is not, then transforming some of these work-paced tasks to self-paced tasks by taping data would be helpful.

6.4 COMMUNICATIONS

Voice communication with earth and, of course, within the cabin will be provided. Frequent vocal communication with earth is essential for morale and for command. The transmission of information required at earth's stations may be supplemented by the human operator who is extremely efficient in this regard. The principal aim is to pass the message with maximum speed and minimum error. The principal hazard is the presence of interfering audio noise, or radio interference.

The techniques for investigating the problems of speech intelligibility are well defined, and the important variables can be readily specified. The basic parameters are:

1. The speech intensity.
2. The speech-to-noise (S/N) ratio.
3. The system gain.
4. The frequency characteristics of speech.
5. The spectrum of the masking noise.
6. The bandwidth of the transmitted signal.
7. The nature of the message set.

The basic requirements for a speech system require a speech-to-noise ratio of 10 db and a bandwidth between 200 and 3000 cps. The primary solution to adequate speech intelligibility involves, on the one hand, boosting the overall speech level. This has limits specified by safety and comfort levels. On the other hand, attenuation of the noise signal can be accomplished by using (1) a helmet, (2) ear pads, and (3) ear defenders. These first two devices can be used to reduce the ambient noise level before it reaches the ear canal, thus raising the S/N ratio. In addition, electronic techniques are available which can be utilized to improve speech intelligibility. These include such operations as peak clipping, automatic gain control, etc.

When the completed estimates of the internal noise spectrum are available, they can be analyzed by a technique based on use of the articulation index (Human Engineering principles for the Design of Speech Communication Systems AFCRC TR 60-27 Aug. 1960) and appropriate design measures taken.

In the event that intelligibility problems do arise, it appears that already existing equipment such as the U. S. Navy system for carrier flight deck communications is capable of providing highly intelligible speech in noise levels of 133 db.

6.5 VISUAL FACTORS

Crew visual requirements for APOLLO cover two separate areas, internal vision and external vision.

6.5.1 Internal Visibility

The elimination of direct visual contact with external high intensity illumination sources or automatic intensity control of external illumination entering the vehicle will simplify internal visual problems. Illumination requirements in terms of intensity, quality and color may be determined in a relatively straightforward manner based on crew task analysis. Tasks requiring visual information which will be required of the APOLLO crew may be divided into four separate classifications: vehicle control, in-flight maintenance, research and reconnaissance, and housekeeping. Vehicle control is considered to include all tasks which are performed using the displays and controls located at the control consoles. It includes communications and other subsystem operations. In-flight maintenance includes surveillance, checkout, and repair of any and all systems and components for which a maintenance capability has been provided. Research and reconnaissance includes lunar exploration, biomedical research and observation, photography and all recording. All remaining functions, including personal hygiene and feeding are included in the housekeeping category. With the exception of maintenance and some housekeeping activities, the foregoing groups of tasks may be allocated to specific areas within the vehicle; this will tend to simplify illumination planning in that a fixed illumination level, based on task requirements, may be provided for each work area.

The following requirements have been established based on an analysis of probable crew functions during an entire mission.

6.5.1.1 ILLUMINATION LEVEL

For normal operation, the illumination level at the control stations should be approximately 40 foot candles. This is a comfortable level for continuous viewing, and operator displays, charts, handbooks, etc. will be designed so that visual acuity requirements, discrimination rates and contrast levels will be compatible. For emergency tasks or short duration tasks, such as in-flight maintenance, a capability of increasing the level to 60 foot candles will be necessary. This may be accomplished by providing portable auxiliary sources which may be directed at any cabin area as required. Most of the research and reconnaissance, and housekeeping tasks will be performed in the mission module which also contains the rest area. For most housekeeping and research tasks, a level of 40 foot candles is adequate assuming again that equipment controls, printed matter, etc., are designed for use at that level. If manual operation of external viewing devices requires some dark adaptation, the overall level in the module may be temporarily reduced or, if this is impractical, a red eyeshield or goggles may be worn by one crew man prior to using the instruments. A separate source and intensity control can be provided in the rest area as well as a shade or curtain to exclude the cabin illumination.

During the course of a normal APOLLO mission, two relatively short periods of high acceleration occur, the first during boost as the vehicle is accelerated to injection velocity and the second near the end of the mission as the vehicle decelerates upon re-entering the atmosphere. Anticipated acceleration levels are such that the crew may be expected to perform manual control tasks if the proper measures are taken to minimize the resulting degradation of human capabilities. The deleterious effect of acceleration stress upon visual acuity has been demonstrated in centrifuge experiments as well as in actual flight situations and considerable data regarding means of offsetting these effects is available. Two remedial steps which may be taken are 1) an increase in illumination level and 2) the special design of those displays which will be used under high G for accurate readout while vision is impaired. In consideration of the continuing experimentation in this area, it is not deemed advisable to attempt to define illumination level or display detail requirements for the high G intervals, but rather to limit current efforts to recognition of the problem and plans to implement its solution during the hardware design phase.

6.5.1.2 VARIATIONS IN ILLUMINATION LEVEL

Light distribution within the operator's normal field of view should be as uniform as possible. Shearer and Downey, in a relatively extensive study of space cabin illumination requirements, recommend that rates of change do not exceed 1.5 percent per 10 minutes of visual angle if crew distraction from this cause is to be avoided. For practical purposes, this rule may be considered as applicable only to those limited areas where precise visual tasks will normally be performed. This includes, display/control panels and consoles, work tables and areas for reading, writing, etc. In other areas, the intensity may be permitted to fall off to a level of 5 to 10 foot candles (with provision for emergency illumination from a portable source).

6.5.1.3 GLARE

Cabin interior surfaces and equipment must be designed to minimize glare. The use of matt finishes wherever possible will optimize contrast levels and help reduce minimum illumination levels. Warning lights which may produce glare should not be located adjacent to critical displays.

6.5.1.4 FLICKER

Flicker rates if produced by any on board lighting must be maintained above detection frequency for the unaided eye. The phase shifting of power supplied to various groups of lamps is recommended to reduce the possibility of stroboscopic effect in rapidly moving displays.

6.5.1.5 COLOR

White light is recommended for both vehicle cabins. This permits the use of the full color range in the coding of displays and controls. It is also acceptable by reason of familiarity for relaxation area illumination. The introduction of color for psychological reasons may be achieved by painting the larger surfaces. Warm white or very light blue is recommended for overhead surfaces, with side walls blue and floor gray to provide some illusion of spaciousness. Light tan or gray is recommended for instrument panels and furnishings which will come in contact with the body. Controls and/or projections to be avoided will, of course, require contrasting colors or shades. The color coding of systems installations for ease of identification will tend to simplify operation and in-flight maintenance. If this system is adopted, the high contrast colors and shades should be allocated to the more critical systems.

6.5.1.6 APPARENT SOURCE

To preserve the illusion of "up", even in the weightless state, it is recommended that all items within a given enclosure or room be oriented so that the areas recognized as the object's top face toward a chosen ceiling. The opposite parallel surface may then be considered the floor and the connecting surfaces the walls. The familiar pattern so achieved may serve to minimize disorientation. To further enhance this effect, it is suggested that the primary illumination appear to emanate from the ceiling to duplicate conditions usually found on the earth's surface where we have learned that most lighting comes from above and shadows are cast accordingly. Another reason for this arrangement is that an optical illusion may result if projecting objects (such as movable controls) are illuminated from below with the resulting shadows cast in an unfamiliar direction. This could contribute to operator confusion and resulting control error.

Among the most recent studies of interior illumination requirements for space vehicles are a Design Study for Cabin Lighting of Orbital Flight Vehicles, dated April, 1960, and a survey of lighting requirements for the APOLLO vehicle by General Electric personnel dated February, 1961.

Both studies considered the same three possible sources of illumination, incandescent, fluorescent and electroluminescent.

The first, incandescent, has the advantages of being flexible and easily tailored to a specific need, of being easily adjustable and directional in nature. Disadvantages of standard incandescent bulbs include relatively low efficiency and low resistance to the vibration environment expected in these vehicles. However, it is possible to design bulbs which are highly resistant to vibration.

The second type investigated was fluorescent. While efficiency is high for these bulbs, installed weight is somewhat higher than incandescent. Furthermore the dimensions of fluorescent lamps are such that little flexibility in arrangement is possible.

Electroluminescent (EL) lighting was also seriously investigated. These panels appear ideally suited for general illumination during launch and re-entry (when the incandescent fixture would be turned off minimizing the possibility of a hot filament failure). However, EL panels cannot reasonably provide illumination at a level high enough to delete the requirement for integrally-lit instruments. Since it became evident that certain staging events would indeed require such instruments (incandescent indicator lights and integrally-lit switches), it becomes academic that the background illumination be other than incandescent as well.

Reviewing the results of this analysis it appears that incandescent light used throughout the vehicle will most reasonably satisfy the illumination requirements; particularly with regard to design flexibility, directionability and control. Furthermore, investigation has indicated that bulbs can be designed and fabricated to operate and survive in the expected APOLLO environment.

With regard to the command module, then, incandescent sources will be provided for integrally-lit instruments. Where applicable, integrally-lit system schematics, switch and indicator lights will also be employed. Finally, background illumination will be provided by incandescent bulbs operating in fixtures located to provide a maximum of 50 foot candles without glare or reflection.

Dimming control will be provided for each main panel as well as background illumination. Care will be exercised in balancing the lighting on each panel (particularly if electroluminescent and incandescent displays are mixed).

In the mission module, overall illumination will be low in order to minimize power consumption, while local spotlights will be provided for food preparation, hygiene and navigation. Capability will be provided here for dimming to extremely low levels to permit dark adaptation, should contact observation be desired.

To develop a suitable lighting system for this vehicle, it is considered mandatory that a lighting mock-up be constructed and used as a design tool. Early in the design of the APOLLO vehicle, the geometry, furnishings, instruments, and controls will be mocked up such that illumination criteria can be empirically established and, subsequently, tested for the design of the individual components as well as the overall cabin. After development of these components this mock-up can again be used to finally qualify the lighting system design.

6.5.2 External Vision

A study of the APOLLO lunar mission profile reveals that the only task-dictated requirements for vision outside the vehicle are associated with the tasks of cislunar and lunar orbital navigation, lunar reconnaissance and re-entry vehicle landing.

Of these, the navigation tasks may be considered to be limited to initial positioning and operation of optics to obtain positive information required for manually directed course correction, and backup calibration procedures.

Lunar reconnaissance operations may be basically visual, with surveillance of the Moon from all angles a primary mission objective. This introduces the opposing requirements for maximum visual freedom under all levels of lunar illumination and maximum protection from direct sunlight as well as from that reflected from the earth and Moon. A viewport or optical viewing device with controllable intensity is therefore required.

Table II-6-II lists the contrast values of sun, earth and Moon based upon a space background of one milli micro lambert as estimated for a point outside the earth's atmosphere (160 km alt.).

TABLE II-6-II

LIGHT SOURCE	LUMINANCE IN LAMBERTS	CONTRAST
Sun	760,000	7×10^{12}
Earth	10	1×10^{10}
Moon	1.2	1.2×10^{9}

It can be seen from this table that an extremely high contrast exists between these celestial bodies and their backgrounds, with values ranging from 10^9 to 10^{12}. This is an order of magnitude that is never observed inside the atmosphere. Unless some filtering or attenuation is provided, it poses a new and severe problem to the astronaut and requires special attention regarding contrast vision and retinal adaptation. With the problem of maintaining a vehicle attitude such that the solar collector is oriented toward the sun, the communications antenna in the direction of the earth, and the viewing device toward the Moon, it seems possible that inadvertent exposure to direct sunlight or to earth glow will occur with resulting loss of valuable time while the astronaut recovers from temporary flash blindness. There may be a requirement, therefore, for an automatic protective device which will function to exclude, or attenuate, high intensity light without seriously compromising the visual reconnaissance capability.

The location and orientation of the viewport or viewing device relative to the vehicle will be critical if maximum viewing flexibility is to be realized. A careful analysis of vehicle position and orientation relative to the sun, earth and Moon, throughout the lunar orbit phase, must be performed to determine view field angles required for lunar observation. The possibility that such an analysis will indicate a requirement for multiple viewing positions must also be considered.

While a strong argument for a viewport is presented from a psychological viewpoint, an even stronger case can be presented for a periscope. Current information indicates that protective devices meeting the response rate and recycling requirements of the APOLLO vehicle are not feasible in sizes large enough to be considered windows or viewports. While this picture may change in the near future, it seems reasonable to consider for the present that Moon reconnaissance will be accomplished using a periscope. Aside from the fact that such an instrument may be designed so that all light passes through a small aperture, small enough for intensity control by devices within the current state of the art, the periscope has the following possibilities to recommend it.

1. Its diameter in the area of the vehicle skin could be small and so minimize the sealing problem.
2. A scanning system which would increase the effective viewing angle is possible.
3. For a small additional weight penalty, a tracking capability which would reduce vehicle attitude control requirements could be built in.
4. Controllable visual angle could be incorporated to occlude direct sunlight or earth glow when either of these bodies is behind the Moon. This would permit lunar observation under minimal illumination while restricting high intensity radiation, since the automatic filter devices would not be required to attenuate the overall light levels.

5. Magnification and reduction capabilities could be provided.

6. It could be integrated with the reconnaissance camera to provide simultaneous viewing and recording and to facilitate accurate camera aiming.

An external visibility problem is introduced by the conventional landing capability of the glide vehicle configuration and by the ground object avoidance capability of the semi-ballistic vehicle configuration. The crew seating arrangement in the semi-ballistic vehicle dictates a remote scanner, viewer arrangement which may best be served by a periscope or by a closed loop television system. The glide vehicle, with its forward facing crew, could incorporate a third solution which calls for removal of a heat shield to expose a windshield.

A television system offers maximum flexibility in installations, view field angle coverage and in its light amplification capability, but suffers by comparison with the passive optical systems when reliability is considered.

For some years, periscopes have been considered as a means of providing external visual information to pilots of high performance aircraft. Studies and flight test programs have been conducted which indicate that flight by periscope is feasible if certain flight path restrictions are observed and if the pilot is trained to respond to the visual cues provided within the relatively narrow field of view.

As noted earlier, the glide vehicle configuration is such that a forward transparency or windshield may be provided. The size and location are such that the view field angle is comparable to simple periscopes.

While the windshield is considered more acceptable, either it or a periscope appears to be satisfactory from a human factors viewpoint and the final choice may hinge on aerodynamic and weight considerations. The windshield does, however, complicate the internal illumination problem by introducing a possible requirement for dark adaptation and self-illuminated flight displays, whereas a hooded eyepiece may eliminate this problem with a periscope.

6.6 RECREATION

The problem of recreation is, psychologically, a matter of providing an adequate level of stimulation as well as sufficient change in the level and nature of the stimulation. As ordinarily employed, the term recreation implies that activity is engaged in by choice, to provide relief from routine tasks. It implies an activity that is desirable and necessary, one which is used to fill otherwise unoccupied time. The necessity for supplying or considering recreation rises from the apparent need for filling unoccupied time and providing a change from routine procedures.

It may be assumed that a minimum of unoccupied time will exist for any crew member. This is probably due to the requirement for obtaining the maximum efficient work output per person. For long periods of time an individual can obviously work effectively for longer than the standard 8-hour day if this is required and he is interested and motivated.

There still may be up to 3 hours of "free time" daily. This could amount to an appreciable portion of time during a long mission. Some special facilities may be required during these periods.

Initially, however, it is essential to study techniques which will reduce the requirements for extra-task recreational facilities by considering the personality of the astronaut, the interactions among members of the crew, the nature of the tasks themselves, and the overall work program. The need for recreation should also be considered in terms of preventing any compromise of adequate mission performance. The "cost" to an individual crew member must be evaluated relative to the demands of the mission, e. g., is it primarily scientific or military, etc.

The psychological makeup of individual crew members will be of primary importance in determining the nature of the recreational facilities which may be required. It appears that the current astronaut may be characterized as follows:

a. High level of general intelligence.

b. Relatively high drive level.

c. Relatively free from anxiety and conflict.

d. Not overly dependent on others; able to tolerate either isolation or close association with others.

e. Motivation dependent on interest in the mission.

f. Ability to tolerate motor inactivity.

In addition, they are individuals who are able to tolerate a variety of stresses. Their adaptability is impressive, as is their strong motivation. They can stand isolation and have a large capacity to withstand frustrations. They are mature, intelligent, action-oriented individuals who spend little time in introspection.

Thus, it is important to realize that the type of individual who will most probably perform space functions is one who is highly-motivated, quite bright, and who is interested in the job itself. To a large degree, "recreation" or pleasure is inherent in the nature of the mission and adequate performance of a task.

6.7 ISOLATION, CONFINEMENT, AND GROUP INTERACTION

The problems of isolation, confinement and group interaction have been treated in some detail in the interim report.

To summarize:

(1) The psychological effects of sensory isolation (deprivation) are powerful, but probably not relevant to the present discussion. Except for lack of stimulation to the graviceptors, the space cabin normally does not involve a drastic reduction in stimuli.

(2) Groups of two or more men performing a mission under adequate leadership survive confinement for weeks at a time without much difficulty. In small groups lonely withdrawal is virtually impossible. Men try very hard not to irritate one another, and to tolerate one another's habits. They throw themselves into work. As long as work is possible, morale is high. The duration of the APOLLO mission is relatively short. There will be a vertical command structure which helps reduce tension. In addition, there will be an optimum combination of official status and maximum skill.

6.8 RESEARCH SUGGESTIONS FOR THE ANALYSIS OF PERFORMANCE UNDER STRESS

1. A standard test package should be developed which is sufficiently flexible to incorporate the principles of performance measurement outlined in an earlier section. The tasks should, of course, be relevant to operational performance requirements. The battery should be used to ask the following questions:

   a. What is the general nature of performance decrement in the several adverse environments or states to which the space travellers may be exposed? These include fatigue, monotony, work-rest cycle, isolation, high and low G, high and low temperature, high and low oxygen partial pressure, high nitrogen pressure, high humidity, and chronic doses of analeptic drugs.

   b. What aspects of performance (e. g., speed, accuracy) are sensitive to these unusual environments?

   c. Where in the system is the breakdown occurring (e. g., motor coordination, cognitive speed, perception)?

   d. What are the synergistic effects of combinations of these adverse conditions ?

   e. What conditions in the work situation maximize or minimize decrement? Will drugs help?

   f. What is the optimal work-rest cycle?

   For the test package, Williams, White, and Lubin have recently made recommendations concerning an apparatus to be used in a centrifuge, or in flight simulators. This test package could be adapted specifically to the APOLLO System. The battery includes a task of the Bekesy type for rapid assessment of visual thresholds, a complex visual discrimination task, self-paced and work-paced vigilance tasks, self-paced and work-paced cognitive tasks, and a task which requires storage and early retrieval of information. Tasks requiring problem solving, decision-making and verbal communication could be added. Such a battery would be programmed with a punched tape, and semi-automatic date processing would be possible.

2. A space cabin simulator should be constructed which provides a shirt-sleeve environment for occupants for two weeks or more. Within the simulator, it should be possible to vary temperature, humidity, composition and pressure of the atmosphere, diet, water supply, spatial orientation cues, time cues, lighting, noise levels, work-load and work-sleep cycles. It should possess equipment which is realistic for the jobs expected of the space flier, e.g., monitoring the environment, monitoring physiological variables (body temperature, pulse rate, etc), psychological variables (diaries, logs, fatigue ratings, etc.), receiving and sending voice and code communications, taking navigational fixes, operating the craft, making cabin repairs, making decisions, solving problems and taking emergency survival action.

The simulator should permit recycling of water and air, and storage of solid waste. All contents should be selected within the framework of minimum weight allowances, miniaturization, and maximum conservation. The space simulator under construction at Brooks Air Force Base appears to meet most of these requirements.

As soon as possible, crews should be selected and put through the various work-rest cycles that have been suggested by Lubin. This will serve three purposes: training, systems analysis, and selection. If a simulated circumlunar flight is carried out with a complete schedule of monitoring realistic displays, rehearsals of launching and re-entry, navigational operations and computations, changes in velocity and attitude of the vehicle, simulated emergencies such as leaks, etc., the capabilities of each crew member would be learned quickly, as would the changes that should be made in the vehicle, in the monitoring procedures, etc. Because of time limitations, probably no two crews will be tested under exactly the same conditions. Presumably flight procedures and vehicle equipment will be under continuous modification. Therefore, each crew, if possible, should act as its own control. The crew should be trained in the simulator under non-flight conditions until stable performance has been achieved. Then the simulated flight should be undergone and if possible, a post-flight set of tests should be given.

The jobs required of the subjects should be as close as possible to the actual jobs expected during the lunar flight. Dials are similar, communication nets are realistic, and so on. The method of analysis is systems analysis. In addition to the simulated jobs, however, the general performance testing battery is aboard, and subjects are given regular practice on it. When a system breakdown occurs, the human factors elements might be analyzed as follows: First, the error or breakdown must be persistent—eg., persistent difficulty orienting the vehicle. Second, it must be reasonably certain that the difficulty is not due to apparatus failure. This negative approach will make it fairly certain that human error is involved. Data from the performance in question should be analyzed to see what aspect of performance may be at fault. The standard performance battery can be used to check on such questions as whether the subject can perceive the signals, read the dials, perform the computations, store and retrieve the data. The next step is to change the system so as to minimize the assumed performance handicap. If this succeeds in eliminating or greatly reducing the error, then presumably there is no need for further analysis. If the procedural change does not succeed, additional hypotheses about the nature of the performance difficulty will be required, and variations of the standard performance battery worked out which will directly test these hypotheses.